U.S. Department of Transportation
Federal Aviation Administration

Advisory Circular

DRAFT

Subject: **AIRFRAME GUIDE FOR CERTIFICATION OF PART 23 AIRPLANES**

Date: 7/6/00
Initiated By: ACE-100

AC No: 23-xx-28
Change:

1. PURPOSE. This advisory circular (AC) provides information and guidance concerning acceptable means, but not the only means, of complying with Title 14 of the Code of Federal Regulations (14 CFR) Part 23 Subpart C and portions of Subpart D. It consolidates the substance of existing Civil Aeronautics Administration (CAA) and Federal Aviation Administration (FAA) letters into a single reference. It also presents information from certain presently existing AC's that cover general topics and specific airworthiness standards. Material in this AC is neither mandatory nor regulatory. AC methods may be freely chosen, or ignored, by an applicant who seeks to demonstrate regulatory compliance. Use of FAA published AC guidance frequently speeds the design approval process for an applicant.

2. CANCELLATION. The following AC's are canceled:

a. AC 23-3, Structural Substantiation of Secondary Structures, dated September 5, 1985;

b. AC 23-4, Static Strength Substantiation of Attachment Points for Occupant Restraint System Installations, dated June 20, 1986;

c. AC 23-5, Cutouts in a Modified Fuselage of Small Airplanes, dated August 6, 1986;

d. AC 23-6, Interpretation of Failure for Static Structural Test Programs, dated September 2, 1986;

e. AC 23-7, Substantiation for an Increase in Maximum Weight, Maximum Landing Weight, or Maximum Zero Fuel Weight, dated July 1, 1987;

f. AC 23-12, Structural Substantiation of Part 23 Airplane Modifications Involving Increased Engine Power, dated January 27, 1993.

3. BACKGROUND. The AC format is current with the airworthiness standards that appear in Part 23 through Amendment 23-51, effective March 11, 1996, and includes all policy in effect as of the ACE-111 Policy Files Index dated June 1, 1994. This information spans approximately 30 years of FAA and CAA letter-written aviation guidance. It includes some historical guidance that dates back to Civil Air Regulations (CAR) 3 and the earlier CAR 04.

4. APPLICABILITY. This material has no legal status. However, to encourage standardization during the certification process, the FAA recommends that the applicant consider this guidance during each small airplane type certificate and supplemental type certificate project.

5. PARAGRAPHS KEYED TO PART 23. This AC includes all the regulatory topics found in Part 23, Subpart C and in Subpart D through CONTROL SURFACES (that is, §§ 23.651 through 23.659).

a. Each AC paragraph corresponds to the applicable Part 23 section for the corresponding amendment shown in the title.

b. Reference to AC information appears without the section “§” symbol, for example, "23.301."

c. Any reference to the like-numbered airworthiness standard is shown with a section symbol, for example, “§ 23.301.”

d. When “Original” appears as the amendment number applicable to a specified section, it specifies that Part 23 of the Federal Aviation Regulations effective February 1, 1965, unchanged by any later amendment, applies to that section. As Part 23 changes are introduced by new amendments, the FAA will make appropriate revisions to this AC.

6. RELATED PUBLICATIONS.

a. Copies of current editions of the following publications may be obtained free of charge from the U.S. Department of Transportation, Subsequent Distribution Office, Ardmore East Business Center, 3341 Q 75th Avenue, Landover, MD 20785: Some of these advisory circulars are also available on the internet at http://www.faa.gov.

AC 20-33B, Technical Information Regarding Civil Aeronautics Manuals 1, 3, 4a, 4b, 5, 6, 7, 8, 9, 13, and 14.

AC 20-44, Glass Fiber Fabric for Aircraft Covering.

AC 20-71, Dual Locking Devices on Fasteners.

AC 20-107A, Composite Aircraft Structure.

AC 21.25-1, Issuance of Type Certificate: Restricted Category Agricultural Airplanes.

AC 23-9, Evaluation of Flight Loads on Small Airplanes with T, V, +, or Y Empennage Configurations.

AC 23-13, Fatigue and Fail-Safe Evaluation of Flight Structure and Pressurized Cabin for Part 23 Airplanes.

AC 23.562-1, Dynamic Testing of Part 23 Airplane Seat/Restraint Systems and Occupant Protection.

AC 23.629-1A, Means of Compliance with Section 23.629, "Flutter."

AC 25.571-1C, Damage Tolerance and Fatigue Evaluation of Structure.

AC 183.29-1, Designated Engineering Representatives Consultant Directory.

FAA Order 8110.4A, Type Certification Process.

FAA Order 8100.5, Aircraft Certification Directorate Procedures.

TSO-C27, Twin Seaplane Floats

NOTE: The information in the above documents is not duplicated in this advisory circular.

b. Refer to 14 CFR Part 1 for the definition of terms.

c. Copies of the following publications are available upon request from the Small Airplane Directorate, Standards Office, Federal Aviation Administration, DOT Building, 901 Locust Street, Room 301, Kansas City, Missouri 64106:

ANC-1(1), *Spanwise Air-Load Distribution*, Army-Navy-Commerce Committee on Aircraft Requirements, 1938.

ANC-1(2), *Chordwise Air-Load Distribution*, Army-Navy-Civil Committee on Aircraft Design Criteria, Amendment –1 dated 3 January 1944.

You may obtain a current edition of the *Guide to Federal Aviation Administration Publications* from the following internet site:

http://www.faa.gov/apa/publicat/GUIDETOC.htm

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Airframe Guidance for Certification of Part 23 Airplanes

SUBPART C—STRUCTURE

GENERAL

23.301 Loads (Amendment 23-48)

(a) Each Basic Loads calculation should be annotated with the airworthiness standard section for which compliance is shown, for example, use 14 CFR Part 23, § 23.301(a); they may also be shown in Design Criteria, Test Plans, Test Reports, and Software Documents.

(b) Cutouts. Doors, windows, access holes, etcetera, in aircraft structure cause redistribution of axial and shear loads, pressure loads, and stiffness changes. Fatigue capabilities may be affected. Damage tolerance capabilities may be affected. Account for all such design changes.

Biplane designs. These airplanes require special aerodynamic load criteria. A previously used typical approach includes:

Developing an equivalent single wing (cellule), which aerodynamically represents the biplane wings. Civil Aeronautics Manual 04 (CAM 04), paragraph 217, provides an acceptable method unless the biplane has an unusual amount of stagger or decalage. Also see CAM 04, Appendix III.

Defining wing and tail loads from Part 23 load conditions using the equivalent single wing.

Distributing the equivalent single wing load between the upper and lower biplane wings. CAM 8, paragraph 212, provides an acceptable method for biplanes with no decalage.

Distributing the individual wing-loads spanwise using a method such as Schrenk or ANC-1. (See Figure 1.)

METHOD	APPROACH TO SOLUTION	LIMITATIONS ON APPLICATION	COMMENTS
ANC-1	Fourier Solution. Based on lifting line theory. Basic and additional lift.	Conventional unswept wings with aspect ratios greater than 5 to 6.	Exact Solution notwithstanding limitations of lifting line theory. Similar Methods: LOTZ GLAUERT (Elements of Aerofoil and Airscrew Theory) ANDERSON (Theory of Wing Sections)
SHRENK'S NACA TM-948	Approximate Solution. Based on lifting line theory. Basic and additional lift.	Conventional unswept wings with aspect ratios greater than 5 to 6.	Accuracy approaches ANC-1 small inaccuracies overshadowed by original assumptions. Much easier to apply.
SHERMAN'S NACA TN-732	Successive approximations to match chord distribution. Each angle of attack requires separate calculation.	Applicable to any straight wing without high lift devices.	Accuracy compares with ANC-1 but unsatisfactory due to separate calculation for each angle of attack.
PLANFORM (Often used for horizontal tail)	Airload distribution directly proportional to chord variation.	Untwisted high taper ratio wings	Unconservative for Taper Ratios < 0.25. Conservative for Taper Ratios > 0.35.
WEISSINGER NACA TM 1120	Iteration process based on modified lifting line theory. Basic and additional lift.	Applicable to straight or swept wings of low or high aspect ratio..	Lifting surface method also presented as applicable to straight wings. Similar to Multhopp Method.
DeYOUNG & HARPER NACA TR 921	Based on simplified lifting surface theory. Basic and additional lift.	Applicable to straight or swept wings of low or high taper ratio.	Extension of Weissinger Method can account for compressibility M > .5 @ S.L.

NOTE: See NACA TN 606 for Empirical Tip Corrections; RM L53B18 for Tip Tank Effects on Load Distribution.

FIGURE 1

Engine power increases. An increase in engine power causes larger loads on, and in, several aircraft structures. A different engine, with different weight, center of gravity (c.g.) and horsepower, will change the inertial, gyroscopic, and aerodynamic loads from those of the previous engine. A changed propeller imparts similar effects on inertial, aerodynamic and gyroscopic loads, which are imposed upon airplane structures. CAM 8 criteria are applicable only to restricted category airplanes certificated under CAR 8 and are not, by themselves, acceptable for compliance with Part 23.

(c) Policy: The FAA discourages attempting ultimate load flight tests unless the applicant understands fully the risks involved; then the FAA advises caution:

See CAUTION notes in 23.305(b) concerning flight tests.

(d) Use of wind tunnel data: Wind tunnel tests may be used to measure a number of parameters used in determining airplane loads. Force models can be used to measure the tail-off and tail-on airplane lift, pitching moment and drag curves, downwash at the tail, and stability and control parameters. Pressure models can be used to obtain the wingspan loading and the pressure distributions on airplane components. Measured pressure distributions are particularly useful in determining loads on secondary structure, including nacelles, canopies and fairings.

Lower performance conventional airplanes are commonly designed without the benefit of wind tunnel testing. Conventional, as used here, means an airplane having a main wing at or near the c.g. and an empennage aft of the airplane c.g. Airplanes with unconventional configurations, unusual aerodynamic features, or high performance airplanes (where compressibility effects cannot be neglected) may require wind tunnel tests.

Scaling of test results to full scale airplane values requires similarity of geometry, Mach Number and Reynolds Number. Similarity of Reynolds Number is the most difficult of these to achieve. Test data acquired at Reynolds Numbers significantly lower than flight may be of little use for certification purposes due to the large corrections that must be applied to the test data. Reynolds Number similarity can be achieved through the use of large-scale models or pressurized wind tunnels. Boundary layer transition strips or dots are frequently used to model flight scale Reynolds Number effects on a model tested at lower Reynolds Numbers.

SPANWISE LIFT DISTRIBUTION

The spanwise distribution of lift on the wing may be obtained from wind tunnel test, analysis, or a combination of analysis and test. The following is a list of commonly used spanwise lift distribution analysis methods.

NOTE: The “Reference” in each item is on the following page of this document (see also Figure 1 of this AC).

1. The National Advisory Committee for Aeronautics (NACA) Technical Report 572 reports on the Anderson method. The wings under consideration covered a complete range of taper ratios and a range of aspect ratios from 2 to 20.

2. Reference 1 (page 228) states that NACA Technical Report 585 contains an exact method.

3. The “Fourier Series Method” is described in Reference 1 on pages 233 to 242. This is the method in ANC-1 (which has tabular forms for ease of calculation). This method uses lifting line theory, which is good for conventional unswept wings with aspect ratios greater than 5 or 6 (see Reference 1, page 247).

4. Weissenger's “Method” Reference 2 is applicable to straight or swept wings of low or high aspect ratio. This is a modified lifting theory method.

5. Schrenk's “Approximation” basically averages the lift forces obtained from an elliptical lift distribution with those obtained from a platform lift distribution. This approximation is very accurate for wings that approach an elliptical platform (Reference 1, page 224). This method is contained in Civil Aeronautics Manual (CAM) 04, Appendix IV. The Limitations in Section 6 state that it applies to the normal range of aspect ratios (from 5 to 12).

6. Reference 3 (page 14) lists NACA Technical Reports 572, 585, and 606. The first two of these technical reports are discussed in Items 1 and 2 above. Technical Report 606 is titled “Empirical Corrections to the Span Load Distribution at the Tip.” Correction is only necessary if the wing is tapered less than 2:1 and has a blunt tip.

7. Reference 4 lists several references: 3.5 through 3.12 (see Enclosure 3).

8. In the paper titled “Application of Microcomputer Software to the Aerodynamic Design of a Motorglider” *Technical Soaring*, October 1993, the lifting line theory FORTRAN program in Reference 5 (pages 159-164) was used (on an Apple Macintosh Plus computer) to calculate the lift distribution of a 17 meter (55.76 foot) wing (aspect ratio unstated). A comparison was made of the lift distribution calculated by Schrenk's approximation.

9. Reference 6 contains a vortex lattice FORTRAN program.

10. Reference 7 is an aeroelastic supplement for the NASTRAN finite element program.

11. NACA TN 3030 ‘A Method for Calculating the Subsonic Steady-state Loading on an Airplane with a Wing of Arbitrary Planform and Stiffness.’ It includes both the aeroelastic effects and the ability to base the span loading on linearized wind tunnel wing section data. It also allows for the correction of wind tunnel model elastics and jig twist.

References

1. D. J. Peery, *Aircraft Structures*, McGraw-Hill, (1950).

2. J. Weissenger, *The Lift Distribution of Swept-Back Wings*, NACA Technical Note 1120, (1947).

3. Anonymous, *Basic Glider Criteria Handbook*, Federal Aviation Agency, (1962).

4. Niu, M. C. Y., *Airframe Structural Design*, Conmilit Press, LTD., (1988).

5. Kuthe, A. M. & Chow, C. Y., *Foundations of Aerodynamics*, Fourth Edition, John Wiley & Sons, New York, (1986).

6. Margason, R. J. and Lamar, J. E., "Vortex Lattice FORTRAN Program for Estimating Subsonic Aerodynamic Characteristics of Complex Planforms," *NASA Technical Note D-6142*, (February 1971).

7. *MSC/NASTRAN Aeroelastic Supplement*, MacNeal-Schwendler Corporation, 815 Colorado Boulevard, Los Angeles, CA 90041-1777.

8. DeYoung, J. and Harper, C. W., "Theoretical Symmetric Span Loading at Subsonic Speeds for Wings Having Arbitrary Planform." *NACA Report No. 921*, (1948).

9. Sivells, J. C., "An Improved Approximate Method for Calculating Lift Distributions Due to Twist." *NACA Technical Note 2282*, (1951).

10. DeYoung, J., "Theoretical Anti-symmetric Span Loading for Wings of Arbitrary Plan Form at Subsonic Speeds." *NACA Report No. 1056*, (1951).

11. DeYoung, J., "Theoretical Symmetric Span Loading Due to Flap Deflection for Wings of Arbitrary Plan Form at Subsonic Speeds." *NACA Report No. 1071*, (1952).

12. Falkner, V. M., "The Calculation of Aerodynamic Loading on Surfaces of any Shape." *ARC Report R&M No. 1910*, (1943).

13. Multhopp, H., "Methods for Calculating the Lift Distribution of Wings (Subsonic lifting-surface theory)." *ARC Report R&M No. 2884*, (1955).

14. Watkins, C. E., Woolston, D. S., and Cummingham, H. J., "A Systematic Kernel Function Procedure for Determining Aerodynamic Forces on Oscillating or Steady Finite Wings at Subsonic Speeds." *NASA Technical Report R*-48, (1959).

23.302 Canard or tandem wing configurations (Amendment 23-42)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

23.303 Factor of safety (Original)

No policy available as of June 1, 1994.

23.305 Strength and deformation (Amendment 23-45)

(a) No policy available as of June 1, 1994.

(b) Intentionally Left Blank

General Comments

Certain FAA Order 8110.4 practices, about returning articles to service that have experienced ultimate load tests, may be relaxed without compromising safety. [Order 8110.4A, currently in effect, was issued more than 23 years after this example situation from Order 8110.4.] For instance, an engine mount assembly can be readily and completely inspected to determine that there is no structural damage (deformation, permanent set, material yielding). The previous FAA Order 8110.4 permits similar practice for limit load tested articles. Exercise judgment to determine which structures can properly be inspected for damage.

The interpretation of a structural failure of a static test specimen has varied greatly on past type certification programs. In the strictest interpretation, if one part (even a rivet) fails beyond limit load but below ultimate load, the test is stopped—the part repaired—and the test rerun. The repair, in this case, becomes part of the type design. In a more liberal vein, a local failure up to ultimate load has been accepted as long as the entire structure being tested was capable of carrying the ultimate load for 3 seconds. The applicant was not required to redesign or structurally "beef up" the locally failed part. In a third interpretation, a specimen was loaded to destruction with a continuously increasing load at a constant rate and with a continuous recording of the test results. The ultimate load was established as the load attained 3 seconds before the maximum load (failure load) was recorded.

In the interest of standardizing interpretations, the following definition of failure is used by all Aircraft Certification Offices to assess the acceptability of a structural static test:

> **Definition:** A structural static failure has occurred when the article being tested cannot sustain an increase in load or cannot sustain the required load for at least 3 seconds. Local failures are allowable if occurrence is beyond limit load and if the article can reach and sustain the required load without failure.

NOTE: The "failure" load observed during the test should be adjusted to account for material variability. Determine the actual strength of the failed structural material(s) and adjust the test results down (or up--the unlikely situation) because the failed material(s) is (are) stronger than the specified value. When a material is chosen according to the FAA airworthiness standards for material strength properties and design values (14 CFR Part 23, § 23.613) actual materials will exceed minimum design values at least 90 percent of the time.

Employ Static Tests

A good design philosophy is to design the structure for no buckling or no occurrence of structural instability below or at limit load. Skin buckling that occurs under load and disappears upon load removal is acceptable. The assessment of a structure at limit load is first a visual check. Deformations may be observed at limit load. However, those deformations should disappear when the load is removed. Also, any deformation that may occur at any load up to limit load should not interfere with safe operation. For example, when static testing a complete wing structure that includes installed control systems, ailerons, flaps, etcetera, the control systems and surfaces should perform their intended function during any deformation that may occur up to and including limit load. The FAA CAUTIONS airplane designers and certifiers to watch out for the SPECIAL EXCEPTION to FAA LIMIT and ULTIMATE load regulatory failure conditions (Euler Column Buckling). COLUMN STRUCTURES, when they are used in a (primary structure) single-load-path design application, cannot be allowed to buckle under either FAA LIMIT or FAA ULTIMATE load conditions. Two common applications of column structures are wing struts and control system pushrods. (See 23.365, Pressurized cabin loads, for additional guidance about this topic.)

Settlement of structure due to the effects of riveting, fasteners, etcetera, does take place during limit load tests. When testing a pressurized fuselage, the pressure differential required by § 23.365 will introduce some settlement in the rivets and fasteners. The differential pressure required is 1.33 times the maximum relief valve setting. For altitudes that exceed 45,000 feet, previously issued Part 23 Special Conditions required a differential pressure of 1.67 times the maximum relief valve setting. Under limit load, visually inspect the test specimen and accept sounds associated with the working of rivets, fasteners, and panels as the applied load increases to limit load. See 04.201, Civil Aeronautics Manual (CAM) 04, Deformations, revised July 1, 1944. (See 23.365 guidance about pressure tests.)

Policy: For metallic structures only, the FAA Small Airplane Directorate allows flight structure to be used as part of an airplane in operational service if the structure has been thoroughly examined, following static tests to any loads, and the structural deformations have remained elastic. That is, the structure should be shown, by proper measurements, not to have experienced material stresses beyond the material yield-point stresses under the applied loads. If the applied static loads equal or exceed the FAA definition of ultimate load, then the test article can be considered to equal or exceed the requirements of Part 23, § 23.305(b). Any deviations to the type design that are created due to structural tests should be dispositioned before the article is subjected to operational flights. If, during any static tests, portions of the structure become visibly damaged, the damaged items should be replaced before the structure being released for operational flight tests. On the other hand, any static test structure that has been tested and has yielded is not a candidate structure for later operational flight tests.

Conduct Flight Tests

In the past, there have been instances where flight tests to dive speed have been accepted as the only means of substantiating secondary structures.

Definition: Secondary structure is not a primary load-carrying member. Failure of secondary structure neither reduces the airframe structural integrity nor prevents the airplane from continued safe flight and landing.

A dive-speed approval does not satisfy the requirement that structure should support ultimate load.

In other instances, dive-speed flight tests have not been accepted as the sole means of showing compliance with the airworthiness standards. In those cases, an applicant presents additional data. Some secondary structure modifications, or alterations, have been approved by structural analyses and tests without dive-speed tests; these include structures like windshields, windows, and radomes.

There have been instances where flight tests to design dive speed (not just to the never-exceed-speed, V_{NE}) have been accepted by the FAA as the only means of "substantiating" secondary structures. This kind of substantiation does not satisfy the requirement for the structure to support ultimate load, nor does it apply the load factors of the V-n diagram to the structure. Flight tests to design dive speed should not be accepted as the sole means of substantiation; the applicant must present additional data to complete the show-compliance-with-Airworthiness-Standards requirements. Certain secondary structure modifications or alterations, which do not change the original external contour, have been FAA approved by structural substantiation alone. Examples of these types of secondary structures are windshields, windows, and radomes.

Caution: Compliance by flight tests should be prudently limited to an 80 percent design flight envelope (V-n diagram; limit loads) until structural tests to all ultimate load conditions are satisfactorily completed (or until all structural analyses to ultimate load conditions are satisfactorily completed).

Caution: Structural flight tests do not necessarily demonstrate ultimate load conditions for secondary structures. Limit dynamic pressure, limit maneuver load factor (and load), and limit landing impact load factor (and load) may be easily achieved during flight tests. Limit gust load factor may be very difficult to achieve during flight tests. It can be dangerous to attempt ultimate load factor (and load) flight tests. Ultimate load flight tests are discouraged.

Information: In rare instances, some secondary structures can be flight tested safely to ultimate load conditions well within the airplane flight envelope (V-n diagram), and also well below the airplane design dive speed (V_D). Landing gear doors are an example of this special case. The landing gear operational speeds, landing gear extended, V_{LE} , and landing gear operating, V_{LO} , can be considerably lower than the airplane design dive speed. Consequently, it is sometimes possible to flight test to an ultimate dynamic pressure for the landing gear that is safely below the limit dynamic pressure for the airplane design dive speed. Angle of attack may be a negligible factor in this case, and maximum airplane yaw angles may be accommodated within the airplane limit flight conditions.

23.307 Proof of structure (Original)

(a) Increases in maximum weight, in maximum landing weight, or in maximum zero fuel weight.

Any one of these changes affects the airplane basic loads and structural integrity and may affect the limitations and performance. Two examples follow:

1. When increasing the original airplane maximum weight, special considerations are necessary. See an acceptable method (below).

2. When replacing a piston engine with a turbo-propeller engine, one must consider that:

    - jet fuel weighs as much as 17 percent more than avgas, and

    - airplane total fuel quantity often must increase.

    Therefore, depending on the location of the fuel tanks, the maximum zero fuel weight might change.

These kinds of modifications should be investigated to verify that either (1) the critical loads have not increased, or (2) the loads that have increased are capable of being carried by the existing or modified structure.

One acceptable method for showing compliance for a weight increase.

Prepare a compliance checklist. It may be an advantage to the applicant to identify the airworthiness standards affected by the proposed weight increase and coordinate that list with the FAA Aircraft Certification Office personnel.

Identify the critical flight, landing, and ground loads. The loads may be obtained from the existing type certificate data (if they are made available to the applicant by the holder of the type certificate) or they can be derived by the applicant. Derived loads should be verified to assure that they produce essentially the same results as those used for the original type certification work.

The airplane structural design airspeeds (see 14 CFR Part 23, § 23.335) should be re-evaluated to determine if the selected airspeeds are adequate at the increased design weights. Lateral gust conditions (see § 23.443) should reflect any changes in yaw moment-of-inertia resulting from revised mass distributions. When pitch and roll inertia affect the airplane loads, due to increasing or redistributed mass, examine these also.

Substantiate all structures affected by load increases, however small. This may be accomplished after the critical loads are identified. Stress analyses, static tests, or a combination of both proof-of-strength methods may be used to substantiate structures capable of sustaining ultimate loads (see §§ 23.307 and 23.641). The FAA encourages, but does not require, the applicant to conduct proof-of-strength tests to both limit and ultimate load conditions—and beyond (for the additional knowledge gained and growth capability). If static tests are used as the proof-of-strength method, the structure should be inspected for detrimental permanent set following the removal of the limit load(s). Any structure that shows detrimental permanent set requires some redesign and retest. When using analyses as the proof-of-strength method, the material yield-point stress should not be exceeded when a limit load is analytically applied.

If an airplane was initially certified with maximum landing weight equal to maximum weight, and if an increase in the maximum weight is applied for, the applicant may take advantage of the five percent difference between landing weight and maximum weight permitted by § 23.473(b). In that case, re-substantiation of the landing gear is not required for the first five percent of the weight increase (as long as the airplane center of gravity (c.g.) remains within the original type certificate limits: See the "weight increases" topic that follows.).

Until Amendment 23-48, Part 23 did not require that a maximum zero wing fuel weight be established (see § 23.343, Design fuel loads). However, for airplane designs with wing fuel tanks, the minimum fuel condition may produce the highest wing-bending moment; it also affects the wing torsional moment. Evaluate these conditions during the showing of compliance phase of the project (see § 23.301).

Verify weight distribution and c.g. design changes. Weight increases or relocated mass items, which change the overall mass distribution, may also change the airplane c.g. at empty weight, maximum weight, and weights in between the two extremes. These kinds of design changes should be carefully investigated for their affects upon the original weight versus the c.g. envelope. The designer should consider the effects of depletable payload items, like fuel; account for c.g. shifts; and calculate the influence these may have upon the whole airplane.

Examine the effects of design changes to the airplane structural damping and speeds. Changes to the maximum weight, maximum zero fuel weight, airplane structural stiffness and the distribution of mass need to be examined with the effects of flutter in mind. Ground vibration survey tests permit the identification of airplane structure nodes, modes, and corresponding frequencies. From these, the airplane flutter characteristics can be analytically estimated. This subject is thoroughly discussed in AC 23.629-1A, Means of Compliance with Section 23.629, "Flutter."

Re-evaluate the fatigue strength or fail-safe strength estimates. A fatigue or fail-safe evaluation should be accomplished if the certification basis airworthiness standards include §§ 23.571 or 23.572, and they should be prudently considered in every design. This evaluation may indicate that cyclic tests should be run on a fatigue test specimen with the modifications incorporated.

The certifier should revise the Airplane Flight Manual and the Instructions for Continued Airworthiness. The revisions or supplements to these manuals should reflect any pertinent changes in weight and balance data, performance, flight procedures, maintenance procedures or practices, life-limited part, etcetera. Note that the Maintenance Manual may also be affected in addition to the Instructions for Continued Airworthiness.

Weight increases: Identify the critical flight, landing, and ground loads. The loads may be obtained from the existing type certificate data if they are made available to the applicant by the holder of the type certificate. The designer (or modifier) can calculate the loads. Verify calculated loads—the older CAM's often show acceptable methods to calculate loads that an airplane designer may use.

Evaluate the airplane design airspeeds (reference § 23.335) to determine if the selected airspeeds are adequate at the increased design wing loads. Lateral gust conditions (reference § 23.443) should reflect any changes in yaw moment-of-inertia resulting from revised mass distributions. An increase in airplane weight frequently causes an increase in wing loading.

Policy: A decision about whether the maximum weight increase is "small" or "substantial" does not affect the application of guidance in this AC.

Wood Airplane Structure

When designing aircraft that contain wooden structures, refer to "Design of Wood Aircraft Structures," Army-Navy-Civil ANC-18 Bulletin, prepared by the Forest Products Laboratory, Forest Service, United States Department of Agriculture, and ANC-23 Panel on Sandwich Construction for Aircraft, Subcommittee on Army-Navy-Civil Aircraft Design Criteria (Aircraft Committee Munitions Board). The following are general interest information items:

In the continental United States, ANC-18 Bulletin, Section 2.1, indicates that 15 percent moisture is considered acceptable for wood used in airplane design. The moisture content expected in service would obviously depend on the geographic region of the earth where the aircraft is operated. However, where the relative humidity is expected to be greater than 90 percent for an extended time (the Tropics), 20 percent moisture content should be assumed. **Note:** As moisture content increases, wood strength decreases.

The FAA knows of no protective coating that will prevent wood from reaching an equilibrium condition in moisture content in ambient conditions.

The FAA recommends that tests be conducted whenever the design is in question.

Specific test requirements for strength due to moisture content, when proof of strength is shown by tests of Civil Air Regulations (CAR) 3.174-1(b), are not outlined and much is left to the judgment of the FAA certification engineer. Test conditions should be reasonable and, without other data, the recommendations of ANC-18 Bulletin should be used (see Section 3.0111). Certification records of four previous successful airplane designs show that the moisture content of the test articles was not documented during tests.

The FAA has no published methods or procedures about the effects of moisture content on the strength of wooden structures. Refer to "Design of Wood Aircraft Structures" (ANC-18 Bulletin) for methods and procedures that are acceptable to the procuring or certificating agency (see Section 1.0 of ANC-18).

Tension Pad Static Tests—Caution

There was a fatal accident involving an airplane that suffered a wing upper-skin failure in buckling. When tension pads are used to apply loads during static tests, they may stiffen thin-skinned structure and bias test results nonconservatively. The following factors should be observed and documented in the test report regarding tension pad use:

Type: round rubber pad with metal back, square rubber pad with metal back, round canvas pad, square canvas pad, etcetera.

Size and number: load simulation dependent, consider the percentage of lifting surface covered (outboard or inboard, fore or aft).

Location: upper or lower lifting surface (is the surface a main tension or compression field?); tension pad proximity to a spar, rib, or stringer structural element; effects of fuel pressure loads.

Agricultural Airplane—Alternate Means of Compliance. (See 23.562, Emergency landing dynamic conditions, for information and guidance about using dynamic seat requirements.)

Substantiation of Secondary Structure

If considering flight tests as a substantiation method, see the topic **Conduct Flight Tests** in 23.305(b).

Failure criteria for secondary structure. When secondary structure experiences forces that vary with angle of attack, or yaw conditions, demonstrate these structures to the same failure criteria as a primary structure. The structure must:

(1) support limit load without detrimental permanent deformation, and

(2) support ultimate load without failure for 3 seconds.

This can often be accomplished using a simple conservative analysis.

Methods

Structural analyses or static tests, or a combination of these, may be used to show compliance with both the limit and ultimate load conditions. The critical points on the flight envelope (V-n diagram) should be examined. The basic loads may be obtained from flight tests, wind tunnel tests, derived data from similar airplane designs, or by conservative analyses. Engineering judgment is involved. Some pertinent considerations include the following:

Wind tunnel basic loads: It may be necessary to apply a conservative factor to ensure the confidence of the FAA project engineer in the full-scale loads. Factors to consider include Reynolds numbers, flow similarities between the tunnel-model and the full-scale airplane, and load measuring methods.

Flight Test Techniques.

- When either compressibility or elasticity, or both, are negligible, 1g flight test data is taken at unique speeds where the angles of attack are equal to the angles of attack calculated for conditions of higher speed and load factor at each critical point on the flight envelope (V-n diagram). Positive values can be taken in level, unaccelerated flight. Negative values are usually taken during roller coaster maneuvers. This method is known as “scaling by dynamic pressure.” These data are then corrected for dynamic and static pressure, and for load factor. These data are used to predict the loads that occur at the same angles of attack, but at higher speeds and load factors.

- If either compressibility or elasticity, or both, are factors, flight test data is taken at various speeds approaching M_D and at various load factors approaching the maximum load factor. These test data can be extrapolated to predict the loads at the critical points on the flight envelope (V-n diagram).

  Flight test techniques such as wind-up turns or roller coasters (push me-pull me) are effective methods to develop load factors above and below 1g. The steady-state nature of wind-up turns makes them a desirable technique to develop positive load factors.

Tests that simulate operational maneuvers can be used to develop loads spectrum needed for design life predictions. These tests should include any approved acrobatic maneuvers and atmospheric turbulence.

When static pressure influences structural loads: The structure experiences a load increase due to the external air load when the external static pressure is different from the internal static pressure. This frequently happens on secondary structures like engine cowls, windshields, and windows.

Also, see 23.301(d) Use of wind tunnel data.

(b) The following proof-of-strength factors should be considered for airplane design changes that may affect structure. (Also, see 23.321, General (flight loads.))

When loads increase, the strength of structure is affected by shear, bending, and torsion—not just one of these conditions.

In some aircraft certifications, the structure was proven by tests only. Any modifications to these aircraft that alter the loads—or the load paths—must be assessed to determine if the change is significant enough to require retests. It is inappropriate to assume that additional strength resides in the structures beyond the values proven by tests.

For stressed-skin wings, if analyses are used to justify the strength between limit and ultimate load conditions, strain-gauge data coupled with panel-buckling stress data may be used to validate that the strength extrapolation is reasonable and correct.

Consider the effects of stress concentration factors.

A small increase in design maximum weight may cause a severe reduction of fatigue life (see 23.572, Metallic wing, empennage, and associated structures).

Any load increase on the wing, tail, or landing gear structures, or passenger, cargo or equipment areas affects the fuselage.

Identify specific materials, dimensions, and processes used in the design (see 14 CFR Part 21, § 21.31, and Part 23, § 23.603).

A previously certificated airplane requires a complete structural proof-of-strength substantiation to the Certification Basis regulations when:

- One puts a turbine engine on a previously certificated reciprocating-engine airplane, or when

- A turbine engine substitutes as a single-power-source for two or more reciprocating engines.

Utilize Structural Analyses

Often, an engineer can perform structural analyses that will substantiate airplane designs and design changes. Contact an engineer who is familiar with the FAA certification process and the particular airworthiness standards. Among others, a Designated Engineering Representative (DER) can sometimes help in this endeavor. This is another way that allows a designer or a modifier to gain FAA approval for changes to the type design. See AC 183.29-1, Designated Engineering Representatives, current edition.

FLIGHT LOADS

23.321 General (Amendment 23-45)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994. (See 23.307 for guidance about airplane weight increases. See 23.471 for guidance about ramp weight and take-off weight.)

(c) No policy available as of June 1, 1994.

For information about Canard and Tandem Wing airplane configurations, see 23.421, Balancing loads for horizontal stabilizing and balancing surfaces.

Laminar Flow aerodynamics information may be found in 23.21, Proof of compliance.

The next two pages, Figure 2 and Figure 3, show a graphical relational image of the airworthiness standards that pertain to airplane loads.

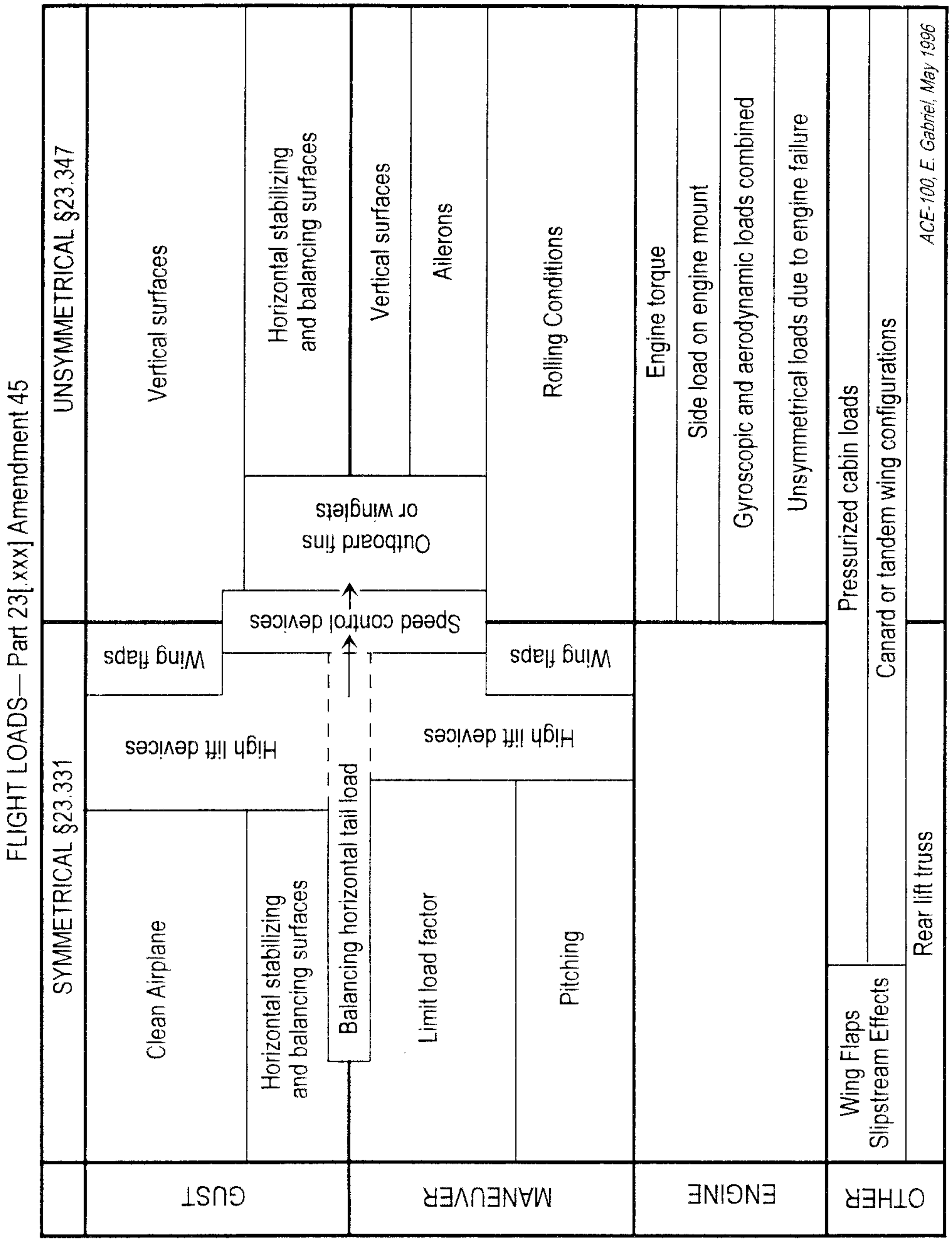
FLIGHT LOADS— Part 23[.xxx] Amendment 45
SYMMETRICAL §23.331
UNSYMMETRICAL §23.347
GUST
MANEUVER
ENGINE
OTHER
Clean Airplane
Horizontal stabilizing and balancing surfaces
High lift devices
Wing flaps
Speed control devices
Outboard fins or winglets
Vertical surfaces
Horizontal stabilizing and balancing surfaces
Balancing horizontal tail load
Limit load factor
Pitching
High lift devices
Wing flaps
Vertical surfaces
Ailerons
Rolling Conditions
Engine torque
Side load on engine mount
Gyroscopic and aerodynamic loads combined
Unsymmetrical loads due to engine failure
Wing Flaps Slipstream Effects
Pressurized cabin loads
Canard or tandem wing configurations
Rear lift truss
ACE-100, E. Gabriel, May 1996

FIGURE 2

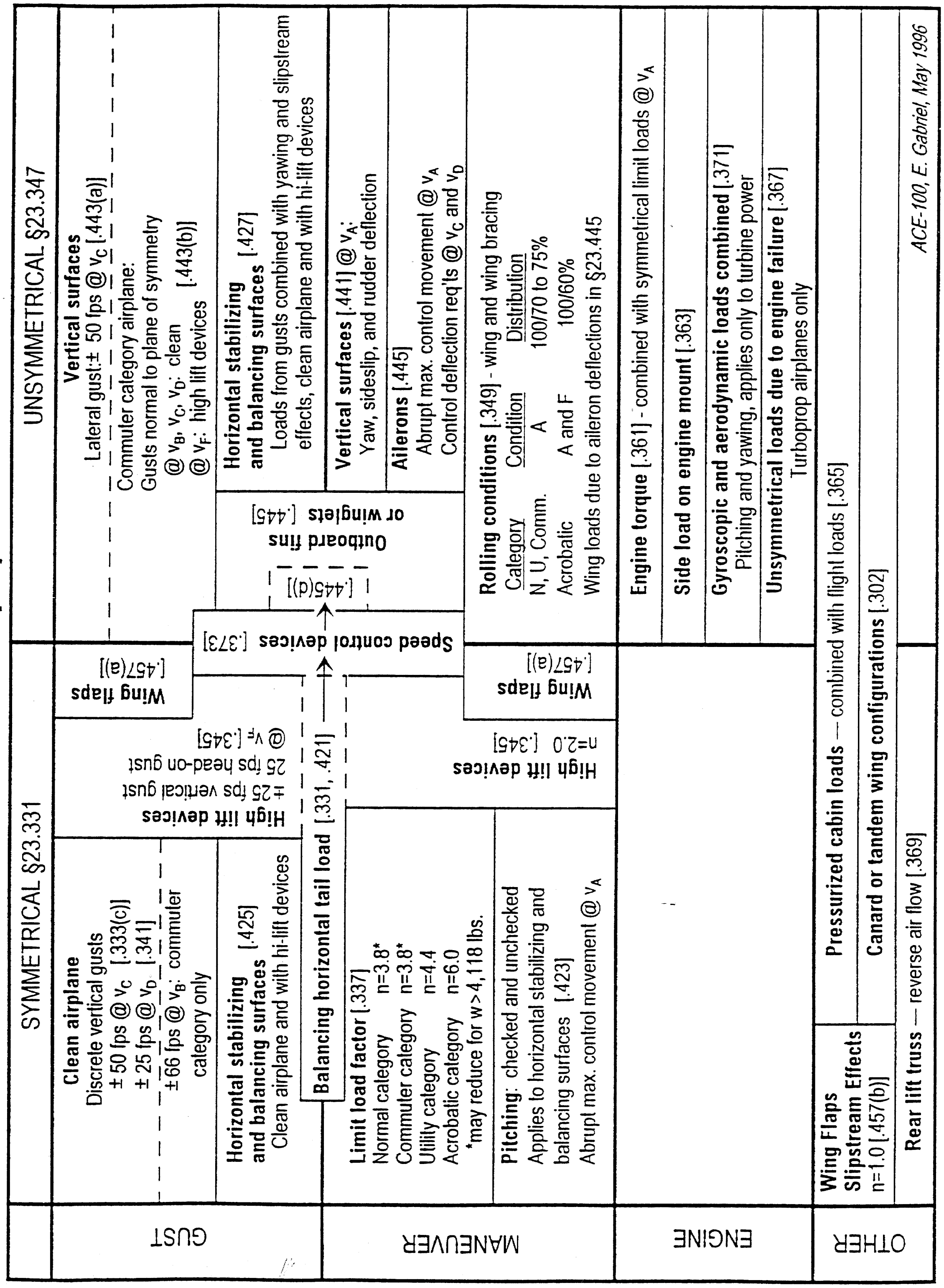
FLIGHT LOADS— Part 23[.xxx] Amendment 45
SYMMETRICAL §23.331
UNSYMMETRICAL §23.347
GUST
MANEUVER
ENGINE
OTHER
Clean airplane
Discrete vertical gusts
± 50 fps @ v_C [.333(c)]
± 25 fps @ v_D [.341]
± 66 fps @ v_B: commuter category only
Horizontal stabilizing and balancing surfaces [.425]
Clean airplane and with hi-lift devices
High lift devices
± 25 fps vertical gust
25 fps head-on gust
@ v_F [.345]
Wing flaps [.457(a)]
Speed control devices [.373]
[.445(d)]
Outboard fins or winglets [.445]
Vertical surfaces
Lateral gust:± 50 fps @ v_C [.443(a)]
Commuter category airplane:
Gusts normal to plane of symmetry
@ v_B, v_C, v_D: clean
@ v_F: high lift devices
[.443(b)]
Horizontal stabilizing and balancing surfaces [.427]
Loads from gusts combined with yawing and slipstream effects, clean airplane and with hi-lift devices
Balancing horizontal tail load [.331, .421]
Limit load factor [.337]
Normal category n=3.8*
Commuter category n=3.8*
Utility category n=4.4
Acrobatic category n=6.0
*may reduce for w>4,118 lbs.
Pitching: checked and unchecked
Applies to horizontal stabilizing and balancing surfaces [.423]
Abrupt max. control movement @ v_A
High lift devices n=2.0 [.345]
Wing flaps [.457(a)]
Vertical surfaces [.441] @ v_A:
Yaw, sideslip, and rudder deflection
Ailerons [.445]
Abrupt max. control movement @ v_A
Control deflection req'ts @ v_C and v_D
Rolling conditions [.349] - wing and wing bracing
Category Condition Distribution
N, U, Comm. A 100/70 to 75%
Acrobatic A and F 100/60%
Wing loads due to aileron deflections in §23.445
Engine torque [.361] - combined with symmetrical limit loads @ v_A
Side load on engine mount [.363]
Gyroscopic and aerodynamic loads combined [.371]
Pitching and yawing, applies only to turbine power
Unsymmetrical loads due to engine failure [.367]
Turboprop airplanes only
Wing Flaps Slipstream Effects
n=1.0 [.457(b)]
Pressurized cabin loads — combined with flight loads [.365]
Canard or tandem wing configurations [.302]
Rear lift truss — reverse air flow [.369]
ACE-100, E. Gabriel, May 1996

FIGURE 3

23.331 Symmetrical flight conditions (Amendment 23-42)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

23.333 Flight envelope (Amendment 23-34)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

23.335 Design airspeeds (Amendment 23-48)

(a) When applying design airspeed criteria**,** the designer may establish a minimum design cruising speed, $V_{C\,min}$, according to the following:

$$33\sqrt{\frac{W}{S}} \leq V_{C\min} \leq 0.9V_H \text{, for normal, utility and commuter category}$$

$$36\sqrt{\frac{W}{S}} \leq V_{C\min} \leq 0.9V_H \text{, for acrobatic category}$$

Both minimum and maximum design cruise speed values, $V_{C\,min}$ and $V_{C\,max}$, may be chosen with the following understanding:

$$V_{C\min} \leq 0.9V_H \text{, and}$$

$$V_{C\max} \leq 0.9V_H$$

(b) No policy available as of June 1, 1994.

(c) The design maneuvering speed is a value chosen by the applicant. It may not be less than Vs $\sqrt{n}$ and need not be greater than Vc, but could be greater if the applicant chose the higher value. The loads resulting from full control surface deflections at V_A are used to design the empennage and ailerons in 14 CFR Part 23, §§ 23.423, 23.441, and 23.455.

V_A should not be interpreted as a speed that would permit the pilot unrestricted flight-control movement without exceeding airplane structural limits nor should it be interpreted as a gust penetration speed. Only if V_A = Vs $\sqrt{n}$, will the airplane stall in a nose-up pitching maneuver at, or near, limit load factor. For maneuvers where $V_A > V_S\sqrt{n}$, the pilot would have to check the maneuver; otherwise the airplane would exceed the limit load factor.

Amendment 23-45 added the operating maneuvering speed, V_O in § 23.1507. V_O is established not greater than $V_S\sqrt{n}$, and is a speed where the airplane will stall in a nose-up pitching maneuver before exceeding the airplane structural limits.

(d) No policy available as of June 1, 1994.

23.337 Limit maneuvering load factors (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

Also, see 23.423, Maneuvering loads.

23.341 Gust loads factors (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

23.343 Design fuel loads (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

23.345 High lift devices (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

(d) No policy available as of June 1, 1994.

23.347 Unsymmetrical flight conditions (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

23.349 Rolling conditions (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

23.351 Yawing conditions (Amendment 23-42)

No policy available as of June 1, 1994.

23.361 Engine torque (Amendment 23-45)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) Section 3.195 (Engine Torque Effects), Civil Aeronautics Manual (CAM) 3, dated June 1, 1958.

This rule requires the design of engine mounts and supporting structure to sustain limit torque at takeoff power and at maximum continuous power, with corresponding propeller revolutions per minute (r.p.m.), for two specified flight conditions.

The rule defines limit torque equal to mean torque multiplied by a factor that is a function of the number of cylinders of a reciprocating engine (see Figure 4):

$$T_{\lim} = F_{cyl} T_{mean}$$

Engine type	Turbo-propeller	Five or more cylinders	Four cylinders	Three cylinders	Two cylinders
F_{cyl}	1.25	1.33	2	3	4

FIGURE 4

Note: The limit engine torque, T_{lim} , is not an engine limit in the sense of 14 CFR Part 33, § 33.7. For structural installation loads purposes, treat it as simply a limit load arrived at by the equation and figure shown. Engine mean torque is available from the engine manufacturer. The engine type multiplying factor, F_{cyl} , is a constant for a given engine without regard to the speed or power at which the engine is operating.

Part 23 Engine Torque Effects

The FAA published engine torque requirements incorrectly in Amendment 23-26. 14 CFR Part 23, § 23.361 was corrected with Amendment 23-45. The incorrectly written rule failed to require the multiplying factor for the torque load. The applied (incorrectly written) rule can result in lower structural loads than previously required from torque loads. These loads affect the engine mount, and either the fuselage or nacelle and wing designs.

Policy: Apply the mean torque factors in the manner that existed in Part 23 before Amendment 23-26 and corrected in Amendment 23-45. Determine airplane design loads for two engine-limit-torques and for two flight-load-conditions. These airworthiness standards for engine torque loads constitute the minimum level of safety required by the FAA for the engine mount, and either the fuselage or the nacelle and wing designs. For airplane designs that have a Part 23 certification basis that encompasses Amendments 23-26 through 23-44, apply the intent of the regulation depicted by the amendments before or after these amendments.

Figure 5 presents a view of torque, aerodynamic, and inertial loads airworthiness standards.

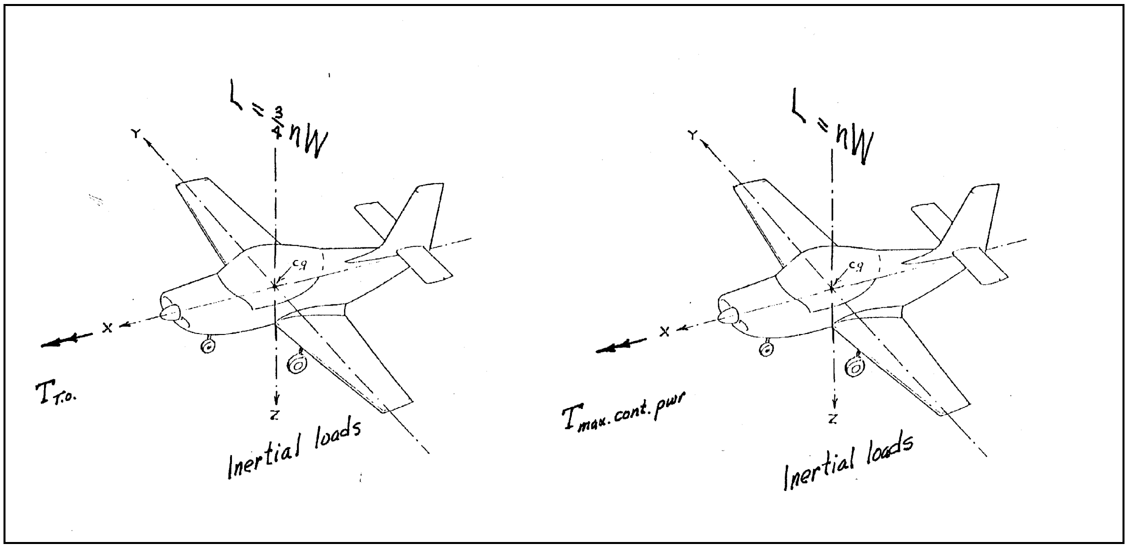

FIGURE 5

23.363 Side load on engine mount (Original)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

23.365 Pressurized cabin loads (Original)

(a) Multiply the maximum differential pressure loads by the 1.5 factor of safety (see 14 CFR Part 23, § 23.303). The maximum differential pressure loads should include the high side relief valve tolerance pressure. Combine them with the ultimate loads of both the normal flight inertia and the local external aerodynamic pressure distribution conditions.

(b) Generally, the aerodynamic pressures can vary, negative to positive, over the length of the fuselage.

(c) No policy available as of June 1, 1994.

(d) For the limit load conditions, airplane fuselage structure should be designed to withstand:

$$P_{\lim} = (1.33)P_{\max.relief.valve}$$

For the ultimate load conditions, combine the 1.5 factor of safety with the 1.33 burst pressure factor to get an ultimate load case for the pressure vessel structural design**.** Part 23 Special Conditions have imposed a 1.67 burst pressure factor for airplanes with design altitudes that exceed 45,000 feet. This practice is consistent with 14 CFR Part 25. Aerodynamic and landing impact loads may be ignored for this load case.

$$P_{ult} = 1.5(1.33)P_{\max.relief.valve}$$

for altitudes that exceed 45,000 feet,

$$P_{\lim} = (1.67)P_{\max.relief.valve}$$

$$P_{ult} = 1.5(1.67)P_{\max.relief.valve}$$

Also, note that the regulation for pressurization tests requires a strength test on a fuselage designed for pressure to the ultimate load condition given by the above equation (see § 23.843).

(e) External door means an opening, a doorway, in the external surface of the airplane. Evaluate door failure effects regardless of whether the door is inward or outward opening. For the purpose of this rule, an emergency exit is a door.

Also, see 23.305, Strength and deformation, under the 'Employ Static Tests' paragraph, for guidance about conducting static tests for pressure vessels.

23.367 Unsymmetrical loads due to engine failure (Amendment 23-7)

(a) The term critical engine means the engine that, when failed, results in the highest structural loads on the airplane.

(b) No policy available as of June 1, 1994.

23.369 Rear lift truss (Amendment 23-48)

(a) Civil Air Regulations (CAR) 3.194 is considered an arbitrary special supplementary reversed airflow condition for design of the rear lift truss, which has no direct relationship to any of the points on the V-n envelope. It has been historically considered as a downwind taxi for a "tail-wheel" type airplane. The lift truss is the brace (frequently a “V,” sometimes “parallel” struts—one to each existing spar) running from the bottom of the fuselage to the lower spar cap(s) of the wing. These lift truss struts usually attach to the wing at about mid-span. “Wing struts” are usually loaded in tension (for positive load factor conditions)—except during negative “g” maneuvers or gusts, inverted flight conditions (aerobatic maneuvers), landing, and taxi. In these latter cases, the struts can be loaded in compression; therefore, they are subject to Euler column buckling phenomena. Even when on the ground, the airplane rear lift truss (or strut) can experience significant compression loads if the airplane has a tail wheel. This is especially true when the airplane is tied down or is taxiing downwind.

(b) No policy available as of June 1, 1994.

23.371 Gyroscopic and aerodynamic loads (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

23.373 Speed control devices (Amendment 23-7)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

CONTROL SURFACE AND SYSTEM LOADS

23.391 Control surface loads (Amendment 23-48)

The control surface load criteria that previously appeared in Appendix B of 14 CFR Part 23 before Amendment 23-42 are not appropriate for the following:

- high-performance Part 23 aircraft,
- aircraft that have spar configurations located aft of the 25 percent chord length, and
- aircraft that have horizontal stabilizer leading edges that are not attached at the fuselage.

The criteria of Appendix B of Part 23 were developed for low-performance aircraft having conventionally designed structure as described in Appendix A of Part 23.

These control surface loads should not be used for tail configurations that contain trim tab or slab-type horizontal-tail surfaces. They should also not be used for T-tail or cruciform-tail (+) configurations where the horizontal tail imposes loads on the vertical structure.

23.393 Loads parallel to hinge line (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

23.395 Control system loads (Amendment 23-7)

(a) Figure 6 presents an overall relational view of control system loads airworthiness standards. Also, see “Control Systems” requirements that are contained in 14 CFR Part 23, §§ 23.671 through 23.701.

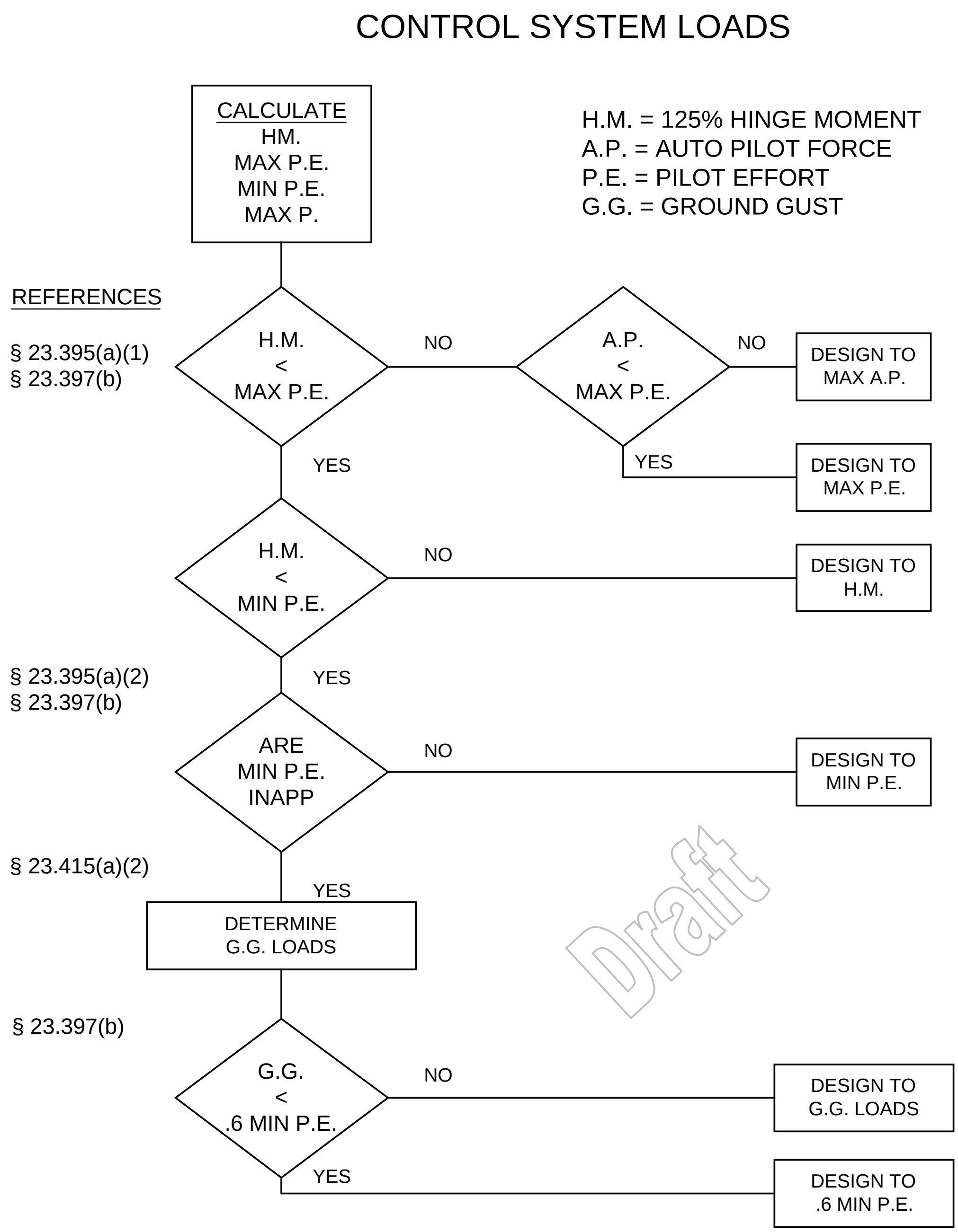

FIGURE 6

(b) See 23.423, Maneuvering loads, for more guidance about control system loads.

(c) The control system loads, and limit control forces, and torque airworthiness standards require that pilot loads be opposed at the attachment of the control system to the control surface horn. For an unconventional multipath control system, which involves control of separate surfaces (each with its own horn), it would be appropriate to expect the pilot forces to be restrained at the control surface horns. If this permits a portion of the system to be designed for less than the minimum pilot-effort forces, special attention should be given to the design of this portion of the system to ensure the rugged system.

Section 23.395 is not applicable to wing flap systems.

23.397 Limit control forces and torques (Amendment 23-45)

(a) No policy available as of June 1, 1994.

(b) Apply the 100-pound load on a single fore-and-aft control wheel anywhere along the periphery or at the tip of the grip-handle on U-type wheels, and in both fore-and-aft directions.

14 CFR Part 23, § 23.397(b) presents both a symmetric and asymmetric 100-pound wheel load for elevator control.

See the accompanying graphic to 23.395 (Figure 6), which presents an overall relational view of control system loads airworthiness standards.

23.399 Dual control system (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

23.405 Secondary control system (Original)

No policy available as of June 1, 1994.

23.407 Trim tab effects (Original)

The following information may be used to account for the maximum out-of-trim condition to consider when determining control surface loads. This rule only applies when maximum pilot-effort forces are imposed upon the control system (refer to 14 CFR Part 23, § 23.397). Then, the trim tab deflection is limited to the maximum out-of-trim condition that can exist under prolonged pilot-effort forces (refer to § 23.143).

> **Note:** This interpretation only applies to a trim tab that is attached to a movable control surface that is further attached to a fixed main-aerodynamic surface (that is, it excludes a trim tab that is attached to a stabilator).

See 23.423, Maneuvering loads, for more guidance about trim tabs.

23.409 Tabs (Original)

No policy available as of June 1, 1994.

23.415 Ground gust conditions (Amendment 23-48)

(a) See the accompanying graphic to 23.395, Control System Loads (Figure 6), which presents an overall view of control system loads airworthiness standards.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

HORIZONTAL STABILIZING AND BALANCING SURFACES

23.421 Balancing loads (Amendment 23-42)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

23.423 Maneuvering loads (Amendment 23-42)

(a) 14 CFR Part 23, § 23.423(a) addresses an artificial load condition intended to provide adequate tail strength for abrupt, unchecked pitch maneuvers up to the design maneuver speed, V_A . The condition is artificial because the airplane will pitch, and the load factor will no longer be 1g, before the elevator reaches full deflection.

A 6g maneuver involves significant airplane pitch velocity, which relieves pilot input, and results in a lower stick force per unit elevator deflection. One may correlate tail angle of attack and airplane pitch velocity with measured and calculated hinge moments to gain confidence in the load calculations.

A flight in which the pilot pulls full-back-stick at design maneuver speed, V_A , and 1g, initially, would likely result in higher elevator angles than would be calculated at 1g, because of airplane response characteristics. However, the tail loads would not likely exceed those calculated at 1g conditions.

The analytical approach is acceptable with the following suggestions:

- For elevator deflection, divide calculated hinge moment by 1.25 (reference § 23.395(b)).
- For an elevator tab configuration, deflect the tab to assist the pilot, per § 23.407, Trim tab effects.

(b) Design maneuvering speed, V_A, (the manner in which it is applied) and the maneuver and gust load distributions are important topics to study and apply correctly.

See 23.421, Balancing loads, for stabilator loads distributions.

Records indicate service failures have occurred due to air loads during flight in severe turbulence or due to unusually severe maneuvers. These occurred in conditions that exceeded the design limitations of the aircraft, as documented by the Civil Aeronautics Board and National Transportation Safety Board accident summaries. Other failures occurred due to material or design deficiencies.

The FAA does not consider the current regulations deficient with respect to the above service experience.

The FAA considers the Part 23 design air-loads requirements generally adequate with the following exceptions:

When rational methods are used to determine horizontal-tail loads, the maneuvering loads pitching acceleration criteria (§ 23.423(b)) can result in nonconservative loads. This could be true for the new high-performance aircraft since the formulas are based on empirical data from old low-performance aircraft. A comparison of Civil Air Regulations (CAR) 3 and Part 23 criteria with National Advisory Committee for Aeronautics (NACA) Technical Report 1007, gives interesting results worth review. Technical Report 1007 is a for sale document available from the Superintendent of Documents, U.S. Government Printing Office in Washington, D.C. Henry A. Pearson, William A. McGowan, and James J. Donegan, wrote the report in 1951. The FAA recognizes that other envelope conditions compensate for the identified deficiency.

The FAA noted a case wherein the negative limit maneuvering load factor (§ 23.337(b)(2)) was substantially exceeded during flight evaluation. There are no specific proposals to resolve these possible problem areas (circa January 1970).

Small airplane designers infrequently use Appendix A to Part 23 to certify airplanes with a gross weight below 6,000 pounds.

Small airplane designers often choose higher design airspeeds and maneuver load factors than the minimum values permitted by Part 23. For the high-performance Part 23 aircraft, a higher design maneuver load factor can usually be used, with little weight penalty, since the airplane (at least the wing) is frequently gust critical rather than maneuver critical. The FAA has no reasons to change the current design airspeeds, maneuver load factor, or gust load factor criteria in Part 23.

23.425 Gust loads (Amendment 23-42)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

(d) No policy available as of June 1, 1994.

23.427 Unsymmetrical loads (Amendment 23-42)

(a) See 23.667(b), Trim systems, for guidance related to non-aerodynamic trim system designs.

(b) No policy available as of June 1, 1994.

(c) A V-tail configuration is considered a "not conventional" design.

T-Tail Design Loads Questions and Answers

Must the airplane demonstrate a flight test maneuver at maximum elevator deflection required in combination with a maneuver at maximum rudder deflection at the design maneuver speed, V_A?

Simultaneous maximum deflection of both the elevator and rudder control surfaces is not required by the regulations, although an airplane designer may choose to design for the combined loads for other reasons. The horizontal and vertical stabilizer maneuver loads regulations require structural substantiation for sudden maximum displacement of elevator and rudder, at the design maneuver speed, V_A, as separate conditions (see 14 CFR Part 23, § 23.423(a) and § 23.441(a)(1)). However, the airworthiness standard for unsymmetrical tail loads, in § 23.427(a), requires that horizontal stabilizer surfaces and their supporting structure withstand unsymmetrical loads arising from maneuver and gust loads, and yaw and slip-stream effects.

If maximum simultaneous rudder and elevator deflection at the design maneuver speed, V_A, is a proposed flight demonstrated maneuver, then this should be a design load condition. Also, note the discussion under the heading of "Additional Considerations" that follows.

Should a symmetrical load on the horizontal stabilizer be combined with a vertical stabilizer load?

This combined load condition is not required by Part 23. Horizontal stabilizer design maneuver and gust loads are normally combined with zero load on the vertical stabilizer. However, the yaw maneuver loads and the lateral gust loads are combined with the horizontal stabilizer balancing load for 1g level flight (see § 23.441 and § 23.443).

Should a 50 feet per second (f.p.s.) gust load applied on the horizontal stabilizer be combined with a 50 f.p.s. gust load applied to the vertical stabilizer?

It is not required to apply a 50 f.p.s. gust load to both the horizontal and vertical-tail surfaces simultaneously. The horizontal-tail gust loads regulation requires that 50 f.p.s. vertical up and down gusts be applied to the horizontal stabilizer at design cruise speed, V_C (see § 23.425(a)(1)). Furthermore, the airworthiness standard for the vertical-tail gust loads requires that a 50 f.p.s. lateral gust be applied to the vertical stabilizer at design cruise speed, V_C, as a separate condition (see § 23.443(a)).

For a T-tail, determine the induced unsymmetrical loads on the horizontal stabilizer when the lateral gust and 1g balance tail loads occur. Examine the resulting combined design load condition to determine if it is critical for the empennage and its supporting structure. Use rational methods to calculate the unsymmetrical loading condition. Also, see heading under "Additional Considerations" that follows.

Using another approach, investigate a 50 f.p.s. gust load at design cruise speed, V_C, for all intermediate angles between horizontal and vertical. If this option is chosen, compare these results to the separate horizontal and vertical loads (previously calculated). Calculate the unsymmetrical loading condition using a rational method.

Note: Combining the horizontal and vertical gust loads is not a mandated FAA requirement that is contained within the regulations; it is a suggestion for a designer to consider.

Additional Considerations

The maneuvers and safe entry speeds proposed for certification should be carefully considered for higher combined loads on the empennage and aft fuselage than would be determined by applying Part 23 requirements. If higher loads are likely to occur, the applicant should perform the additional investigations.

It is unacceptable to determine the horizontal stabilizer unsymmetrical load from § 23.427(b)(1) and (2) for a T-tail airplane. This formula first appeared in Civil Air Regulations (CAR) 03.2214, effective November 13, 1945, long before the T-tail configuration came into use. This rule was recodified into Part 23 as it first appeared in CAR 03. With Amendment 23-14, effective December 20, 1973, the FAA added a proviso stating that the formula may be used "in the absence of more rational data for airplanes that are conventional in regard to location of engines, wings, tail surfaces, and fuselage shape." (See § 23.427(b).) [Conventional means the airplane's center of gravity (c.g.) is located within the boundaries of the wing mean aerodynamic chord with the engine(s) forward and the empennage aft. Also, the horizontal stabilizer touches the fuselage with the vertical stabilizer above (not including T, V, + or Y configurations).]

AC 23-9, Evaluation of Flight Loads on Small Airplanes with T, V, +, or Y Empennage Configurations, provides guidance about this topic. **NOTE: If the**

diagrams are to be reproduced, or the AC revised, the “roll axis and moment reference” and the “moment reference” points should be removed.

Tail configurations beyond the scope:

Quartering lateral relative wind generates aerodynamic loads on both the horizontal stabilizer and the vertical stabilizer. Airplane yaw maneuvers combined with balancing tail loads or airplane pitch maneuver loads contribute to these aerodynamic conditions. For an empennage configuration where the vertical stabilizer structure supports the horizontal stabilizer (T-tail or cruciform (+) configurations), the horizontal stabilizer aerodynamic loads add to the aerodynamic loads that exist on the vertical stabilizer. In addition to their contributions to fuselage torsion and fuselage bending in two planes (horizontal and vertical), they increase the vertical stabilizer structure loads (bending, tension, and compression). For example, the yaw maneuver conditions produce vertical stabilizer surface air loads that generate unsymmetrical air loads on the horizontal stabilizer (see § 23.441). One should combine the vertical stabilizer side load with the unsymmetrical horizontal stabilizer balance load (for 1g level flight), plus any slip-stream effects, to realistically estimate the vertical stabilizer bending and fuselage torsion load conditions. The diagrams shown in AC 23-9, Figure 1, demonstrate how the lateral load on the vertical stabilizer in a sideslip maneuver (also rudder deflection or lateral gust) influences the pressure distribution on the horizontal stabilizer. The diagrams also illustrate how a roll maneuver is more critical for a conventional empennage configuration.

In summary, the Part 23 regulations do not require the designer to combine the symmetric pitch maneuver loads with the yaw maneuver loads (see § 23.423 and § 23.441). However, lateral gust and yaw maneuver loads should be combined with 1g level flight loads.

V-tail and Y-tail design loads: AC 23-9, paragraph f, provides guidance about control surface and system loads for airplanes with control surfaces that receive simultaneous inputs from more than one control axis.

For acrobatic category airplanes, which are intended to perform “flick” or “snap” rolls, the unsymmetrical loading on the horizontal stabilizer should be calculated using conservative assumptions.

Instrumented flight-test results may be used instead of conservative assumptions.

VERTICAL SURFACES

23.441 Maneuvering loads (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

Policy: Do not use Appendix B of 14 CFR Part 23 to determine control surface loads. See 23.391 Control surface loads, for additional guidance about Appendix B of Part 23. Additionally, avoid misusing information that appears in Civil Air Regulations (CAR) 3.219; a tail torsional moment that is adequate for a stabilizer with a main spar at the quarter-chord may be inadequate if the main spar is located nearer mid-span. Part 23, Appendix A, as amended by Amendment 23-48, provides helpful guidance.

23.443 Gust loads (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

Policy: Do not use Appendix B of 14 CFR Part 23 to determine control surface loads. See 23.391**,** Control surface loads, for additional guidance about Appendix B of Part 23.

23.445 Outboard fins or winglets (Amendment 23-42)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

(d) No policy available as of June 1, 1994.

AILERONS AND SPECIAL DEVICES

23.455 Ailerons (Amendment 23-42)

(a) Differential-deflection ailerons are not subject to 14 CFR Part 23, § 23.459, Special devices. Instead, apply the airworthiness standards for § 23.455, Ailerons, and § 23.683, Operation tests.

Previously generated data may be used.

(b) No policy available as of June 1, 1994.

Policy: Do not use Appendix B (to Part 23 before Amendment 23-42) to determine control surface loads. See 23.391**,** Control surface loads, for additional guidance about Appendix B of Part 23.

23.457 Wing flaps (Amendment 23-48) [Removed]

23.459 Special devices (Original)

Differential-deflection ailerons are not subject to 14 CFR Part 23, § 23.459, Special devices. Instead, apply the airworthiness standards for § 23.455, Ailerons, and § 23.683, Operation tests.

Previously generated data can be used.

Dive-brakes are subject to this airworthiness standard (§ 23.459).

GROUND LOADS

23.471 General (Original)

Use a 50-50 load distribution on the forward wheels of a four-wheel landing gear on amphibious airplanes with twin seaplane floats. Also, assess the safety characteristics of the float configuration regarding the effects of float deflections and the skidding action of a float.

Figure 7 presents an overall view of landing gear airworthiness standards.

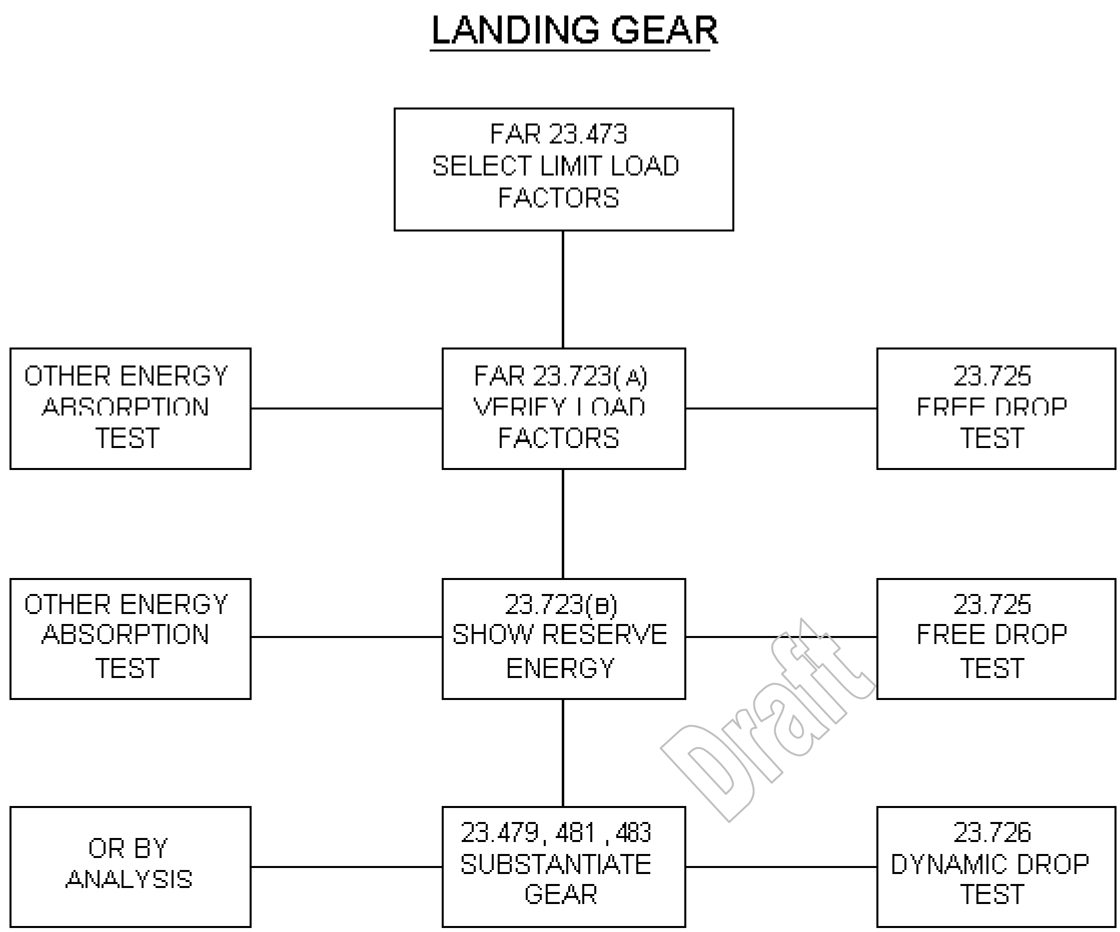

FIGURE 7

Ramp weight versus takeoff weight: In the late 1970's, only one applicant sought an exemption for ramp weight: The jet airplane was certified under Civil Air Regulations (CAR) 3 airworthiness standards. The maximum ramp weight increment became the average weight of fuel consumed during 12½ minute pretakeoff operations at ground idle power. This permitted up to 250 pounds excess fuel weight for ramp weight, which, incidentally, is 2 percent of the maximum takeoff gross weight. Other applicants apparently gained approval for a different weight increment—the amount of fuel weight consumed during 5 minutes ground operation at takeoff power. While this second criterion does not appear conservative, the increment amounted to less than 1 percent of the maximum permissible flight weight.

Ramp weight increments above takeoff gross weight should be restricted to the range of values suggested by previous approvals. Additionally, the ramp weight should be considered the design maximum weight for ground load conditions and assumptions (see 14 CFR Part 23, §§ 23.473 and 23.321(b)(2), or their predecessor regulations). Small weight increments like these may be justified by analyses when one uses the original certification loads, analyses, and drop tests.

All later FAA approvals should accurately inform the pilot about aircraft gross weight before takeoff. Show the maximum ramp weight as a limitation on the Type Certificate Data Sheet and in the Airplane Flight Manual.

Policy: For maximum ramp weight (taxi weight), provide the pilot a means to accurately determine the aircraft gross weight at brake release for takeoff.

23.473 Ground load conditions and assumptions (Amendment 23-48)

(a) Ramp weight and takeoff weight: Ramp weights above 12,500 pounds and ramp weights in excess of the airplane maximum takeoff weight can be utilized for small airplanes and still retain the 14 CFR Part 23 certification basis. To utilize a ramp weight in excess of the maximum takeoff weight, apply the following guidelines:

The difference between ramp weight and takeoff weight should be limited to the weight of fuel that can reasonably be burned off by the engines from start-up to the point of initiating the takeoff roll. Fuel burn-off considerations will include engine start(s), taxi to the runway, engine run-up, and taxi to the initial takeoff point on the runway. Also, for the specific airplane, make these estimates for an average size airport that the airplane will likely operate out of during its operational use.

Since airplane takeoff performance will be based on the maximum takeoff weight, the amount of fuel used for the takeoff run should not be counted as part of the difference between ramp weight and takeoff weight.

Policy: The increment of weight above maximum takeoff weight should be limited to small values consistent with the above-noted reasoning and, generally, will not exceed 1 percent of the takeoff weight.

Provide the pilot an accurate means to determine the airplane gross weight at the takeoff condition just before brake release.

The airplane design should comply with § 23.485 through § 23.511, excluding § 23.511(b) and (c)(1), with ramp weight substituted for maximum weight. These airworthiness standards address:

- Side load conditions;
- Braked roll conditions;
- Supplementary conditions for tail wheels;
- Supplementary conditions for nose wheels;
- Supplementary conditions for skiplanes;
- Jacking loads;
- Towing loads; and

- Ground load; unsymmetrical loads on multiple-wheel units.

Two limitations are involved:

(1) Ramp weight is limited by airplane structural integrity under ground loads, and

(2) Maximum takeoff weight is limited by structural integrity or airplane performance.

The Airplane Flight Manual (AFM) should clearly present these limitations to the pilot.

Display maximum ramp weight limitations in at least two places—the Type Certificate Data Sheet and the AFM.

(b) Obtain the correct fuel capacity and fuel weight to show compliance for design landing weight (reference § 23.473(b)(2)) by using:

- the entire airplane fuel capacity (including unusable and residual fuel), with

- the maximum appropriate fuel densities shown in the Note found in the Powerplant Guide for the Certification of Part 23 Airplanes (AC 23-16), 23.955 Fuel flow, (c) Pump Feed Systems, paragraph (3).

For avgas, use 6.0 pounds per gallon.

(c) Before Amendment 23-7, § 23.473(c) referred to the requirements of 25.1001 for fuel jettison requirements.

Warning: The design landing weight of a Part 23 airplane may be less than that allowed by § 23.473 when a fuel jettisoning system is installed and the one engine inoperative rate-of-climb requirements of § 23.67(a) are met.

The other requirements related to this subject are so different in Parts 23 and 25 that the jettisoning requirements of current Part 25 are not compatible with Part 23. These differences include structures and loads, aircraft performance, and accounting for temperature and altitude effects.

Therefore, the FAA did not intend the jettison requirements of Part 25 (Amendment 25-18) to be applied to Part 23. The preamble of Amendment 23-7 states that amended § 25.1001 does not now reflect the appropriate requirements for Part 23. Amendment 25-18 was intended to cater to the needs of airline transport airplanes and to preclude the need for additional exemptions from the pre-amendment jettisoning requirement.

For these reasons, the application of Part 25 jettisoning requirements to a Part 23 certification is not considered acceptable.

(d) No policy available as of June 1, 1994.

(e) No policy available as of June 1, 1994.

(f) No policy available as of June 1, 1994.

(g) No policy available as of June 1, 1994.

Information about an Increase in Maximum Weight, Maximum Landing Weight, or Maximum Zero Fuel Weight may be found in 23.307, Proof of structure.

It is acceptable to show compliance with §§ 23.485, 23.493, and 23.499 at design landing weight (Side load conditions, Braked roll conditions, and Supplementary conditions for nose wheels). Other ground load conditions, which address ground handling or taxiing conditions, should be substantiated to the design maximum weight.

Seaplanes, Float planes, and Amphibians

Separate weight limits are acceptable for amphibians for operation on land and on water because separate certification criteria exist within the regulations.

For an amphibious airplane design, the following ground load conditions and assumptions are appropriate (see § 23.473, paragraphs (e), (f), and (g)) when the design landing weight is less than the maximum weight (see § 23.473 (b)):

- § 23.473(d)—Determine the landing descent velocity from

$$V = 4.4\left(\frac{W}{S}\right)^{1/4},$$

- § 23.473(e)—Assume a wing lift not exceeding two-thirds of the airplane weight exists throughout the landing impact, and

- § 23.473(f)—Energy absorption tests.

It is inappropriate to use a landing weight less than the design maximum weight in § 23.473(g), since the airplane inertial load factor may not be less than 2.67. Correspondingly, the ground reaction load factor may not be less than 2.0 at design maximum weight.

These conditions do not address the question, how much can the gross weight be increased by analysis of prior drop tests without actually conducting a drop test at the increased weight? Any answer should consider all the factors, such as remaining shock-strut travel, metering pin design, energy absorption characteristics (area under the accelerometer time-history trace), etcetera, and how those factors change at the increased gross weight configuration. The need to conduct or ignore drop tests should be reviewed cautiously.

Policy: A decision about whether the maximum weight increase is “small” or “substantial” does not affect the application of guidance in this AC.

Information about an Increase of Design Landing Weight in Special Federal Aviation Regulation No. 41 (SFAR 41) Airplanes

See AIR-100 Memorandum entitled, "**ACTION:** Certification Procedures for Type Design Changes to SFAR 41 Airplanes" (Figure 8).

Subject: **ACTION**: Certification Procedures for Type Design Changes to SFAR 41 Airplanes

Date:

From: Acting Manager, Aircraft Engineering Division, AIR-100

Reply to Attn. of:

To: All Aircraft Certification Directorates
All Aircraft Certification Offices

This memorandum provides procedures necessary to establish the certification basis for changes to the type design of airplanes previously type certificated under Special Federal Aviation Regulation 41 (SFAR 41). Because of the termination of SFAR 41, the regulations incorporated by reference in the type certificate are no longer appropriate for approval of changes to the type design of these airplanes. This memorandum supersedes previous policy regarding changes to seating capacity and weight increases on SFAR 41 aircraft.

Special Federal Aviation Regulation 41 was issued based on the premise that it would be terminated when suitable airworthiness regulations for that category of airplane could be developed. This regulation was terminated on September 13, 1983, thus ending the effectivity of the regulation for an application for an amended or supplemental type certificate (STC). However, type certification approvals that have been issued under SFAR 41 remain effective for the life of the type certificate (TC). The termination of SFAR 41 was not intended to freeze the design of the airplanes already type certificated; it was intended that the exceptions provided by SFAR 41 be terminated. However, the regulatory actions did not provide for changes to airplanes already type certificated under SFAR 41.

Because SFAR 41 has been terminated, the associated requirements may no longer be used for purposes of issuing amended TC’s and STC’s. For this reason, SFAR 41 is very unique among regulations that support older aircraft, such as the Civil Air Regulations.

<u>BACKGROUND</u>.
Special Federal Aviation Regulation 41 was a temporary rule intended to increase aircraft availability for the commuter market, which was burgeoning since enactment of the

FIGURE 8 (Page 1 of 5 Pages)

Airline Deregulation Act of 1978. This regulation was intended to provide relief to the industry and the public from the lack of suitable certification procedures and standards, and to bridge the gap between Title 14 Code of Federal Regulations (CFR) parts 23 and 25 for the type and size of airplane that is appropriate for the commuter category. By meeting the SFAR 41 provisions, these airplanes could accommodate the surge in commuter airlines immediately following deregulation.

This regulation permitted small propeller-driven multi-engine airplanes to carry more passengers than that permitted under 14 Code of Federal Regulations parts 1, 21, and 23. It provided exceptions to permit an increase in both the maximum certificated takeoff weight and the seating capacity of certain airplanes intended for commuter airline use that were type certificated under part 23. Additional airworthiness standards were imposed to enhance safety of the airplanes.

The commuter category regulations were incorporated into part 23 under amendment 34, dated January 1987, and provided the standard regulations for commuter category airplanes. This amendment incorporated the requirements of SFAR 41. It also mandated the International Civil Aviation Organization's (ICAO) requirements that previously were optional. Additionally, the requirements of Appendix A of part 135, and a few other requirements concerning cabin safety issues were included.

SFAR 41 combined additional special airworthiness requirements with referenced part 25 airworthiness and 135 operating requirements as its regulatory basis. When SFAR 41 was terminated, the airworthiness standards of SFAR 41, including those applicable part 25 standards, were incorporated into part 23 amendment 34 for commuter aircraft, and the operating requirements were incorporated into the appropriate operating rule. Hence part 23 amendment level 34 (which incorporated the part 25 or applicable reference) is the earliest amendment level that can be used for modifications that are affecting the areas listed in SFAR 41. In keeping with this concept, the following guidance will emphasize addressing those areas (which in turn reference the particular requirement) which are affected by SFAR 41, such as Landing Performance, Fatigue Evaluation of Flight Structure, Doors and Exits, Cowlings, Compartment Interiors, Landing Gear, Fuel System and Components Crashworthiness, Shutoff Means, Fire Detector Systems, Fire Extinguishing Containers and Agents, and Fire Extinguishing Materials.

Over recent years, the levels of safety associated with the commuter category regulations have increased from that of SFAR 41, based on the small airplane regulations, to that which are very close to the transport category regulations. The basic premise behind the current policies for the procedures and airworthiness standards for alterations to SFAR 41 airplanes is that the highest possible degree of safety in the public interest, should be achieved. In dealing with this premise, the FAA continually weighs the desire for the highest level of safety with the cost to the manufacturers, operators, and the

FIGURE 8 (Page 2 of 5 Pages)

traveling public for achieving that highest possible degree of safety. Based on this premise, this memorandum will provide certification procedures for alterations of SFAR 41 airplanes.

CERTIFICATION PROCEDURES.
These procedures are applicable for changes to airplanes previously type certificated in the commuter category under SFAR 41, and provide the necessary additional guidance when following the normal procedures specified in part 21. This certification basis is progressive and is correlated with the extensiveness of the change and the effect the change has on cabin safety, flight performance, and flight characteristics. This certification procedure includes the policy of applying airworthiness standards based on the degree of the extensiveness of the change and the effect that the change has on the airworthiness of the airplane.

Due to the termination of SFAR 41, a change to an airplane type certificated under SFAR 41 is handled differently than other products. The SFAR 41 requirements incorporated by reference in the TC of such an airplane have expired, and may no longer be used for purposes of issuing amended TC's and STC's. However, those portions of the certification basis consisting of the part 23 regulations should be usable for repairs and small minor changes.

a. Changes to type certificated products are accomplished in accordance with § 21.101. The certification basis must ensure compliance with § 21.101 for airplanes type certificated under SFAR 41. Section 21.101(a) allows an applicant to comply with the regulations referenced in the type certificate or the regulations in effect at the date of the application for the change. In keeping with this concept, and because there is no longer a certification basis for SFAR 41 airplanes, the airworthiness standards of part 23 that are referenced as part of the original type certification basis and are not affected or related to SFAR 41 remain appropriate for the regulations referenced in § 21.101(a)(1). Further, for those areas not affected by SFAR 41, when determining the certification basis, the procedures of Order 8110.4A Type Certification Process, Paragraph 14.c., Changed Aviation Products, are applicable.

b. For those areas affected by SFAR 41 (i.e. Landing performance, Fatigue Evaluation of Flight Structure, Doors and Exits, Cowlings, Compartment Interiors, Landing Gear, Fuel System and Components Crashworthiness, Shutoff Means, Fire Detector Systems, Fire Extinguishing Containers and Agents, and Fire Extinguishing Materials), the applicable requirements will be the current certification regulations at the date of application for the change. Exceptions to this policy are contained in paragraph d.

c. In addition to complying with the airworthiness standards specified in the certification basis, the altered airplane will have to comply with other applicable

FIGURE 8 (Page 3 of 5 Pages)

requirements, such as the requirements listed in the operating regulations under the provisions of Title 14 CFR parts 91, 121, or 135, that are applicable to the change. The following additional regulations are also applicable:

(1) Special conditions deemed necessary under § 21.16;

(2) Equivalent levels of safety findings in accordance with § 21.21;

(3) Applicable noise requirements of 14 CFR part 36;

(4) Applicable fuel venting and emission requirements of 14 CFR part 34; and

(5) Exemptions in accordance with 14 CFR part 11.

For example, an increase in the maximum takeoff weight is considered an acoustical change that must be evaluated to the requirements of 14 CFR part 36.

d. The following exceptions, based on existing policies, are considered applicable for the regulations affected by SFAR 41 requirements.

(1) An applicant for a change may demonstrate compliance with earlier regulations, but not earlier than the regulations incorporated in part 23 Amendment 34, if the effect of the proposed change is non-significant, taking into account earlier design changes and previous updating of the type certification basis.

(2) An applicant for a change may demonstrate compliance with earlier regulations, but not earlier than the regulations incorporated in part 23 Amendment 34, if compliance with a regulation in effect on the date of application would not contribute materially to the level of safety of the product to be changed, or would be impractical.

(i) Compliance with the later amendment would be considered to "not materially contribute to the level of safety" if the level of safety achieved by the existing design with the proposed change would not be enhanced by compliance with that later amendment. To demonstrate, the applicant should show that the level of safety achieved by the existing design incorporating the proposed design change would achieve a safety level similar to that reflected in the later amendment. Evaluation factors to be used for the assessment should include:

(a) A clear understanding of the regulatory change and what prompted the change;
(b) A detailed knowledge of the proposed design feature; and
(c) A comprehensive review of the applicable service experience

FIGURE 8 (Page 4 of 5 Pages)

(ii) Compliance with the later amendment would be considered “impractical” when the applicant can establish that the resource requirements of the design change and related changes necessary to demonstrate compliance with the amendment would not be commensurate with the resultant safety benefit. Where compliance with the later amendment would prompt a redesign, the resources of redesigning other parts of the product to accommodate this redesign also would be considered.

e. For modifications in areas that are affected by SFAR 41, changes to the type design will be documented in the type certificate data sheet or the STC continuation sheet. This documentation will include the following statement-

“This modification meets the commuter category requirements as follows:”

Regulation	Amendment level
23.xxx	Amendment 23-xx
23.xxx	Amendment 23-xx
“ “	“ “ “

f. Because design changes vary in complexity and magnitude but may have a cumulative effect in regards to the commuter regulations, each application for a change to an SFAR 41 airplane must be evaluated and certification basis established on a case by case basis working in conjunction with the Standards Staff of the Small Airplane Directorate.

Abbas A. Rizvi

Attachment-
Appendix

FIGURE 8 (Page 5 of 5 Pages)

23.477 Landing gear arrangement (Original)

No policy available as of June 1, 1994.

23.479 Level landing conditions (Amendment 23-45)

(a) No policy available as of June 1, 1994.

(b) The drag component for landing gear loads may not be less than 25 percent of the design load factor multiplied by the gross weight of the airplane. This equation is observed in Appendix C of 14 CFR Part 23 under main wheel loads. This does not mean that a separate drop of the main gear, at maximum gross weight, is required to obtain the load factor used in making this calculation.

Two different incline plane angles are given in the Civil Air Regulations (CAR) and Title 14 of the Code of Federal Regulations (14 CFR).

> Approvals have been granted using CAR 3, before May 3, 1962, which defines the inclined plane angle to be ARC tan K for the level landing case with the nose wheel just clear of the ground.
>
> In CAR 3 (Amendment 3-7), dated May 3, 1962, the regulations show the inclined plane angle to equal ARC tan [nk/(n-L)] (see CAR 3.245(b)(2), Figure 3-12(b), and § 23.479(a)(2)(ii), Appendix C of Part 23).
>
> Manufacturers of amphibious floats for aircraft certificated under Part 23 have used inclined plane angles as ARC tan K.

Policy: CAR 3 landing gear drop test data substantiated to requirements dated before May 3, 1962, are not acceptable for aircraft certificated to CAR 3 or 14 CFR Part 23 on or after the 1962 date.

(c) No policy available as of June 1, 1994.

(d) No policy available as of June 1, 1994.

23.481 Tail down landing conditions (Original)

(a) No policy available as of June 1, 1994.

(b) 14 CFR Part 23, § 23.481(b), Tail down landing conditions, reads as follows:

> "For airplanes with either tail or nose wheels, ground reactions are assumed to be vertical, with the wheels up to speed before the maximum vertical load is attained."

A primary concern about the last phrase of the above sentence is that it can be interpreted at least two ways (both of which are correct):

1. Since the wheel is assumed up to speed, the drag load caused by wheel spin-up is zero when the maximum vertical load occurs; or

2. Maximum wheel spin-up and spring-back are not zero but occur before the landing gear maximum vertical load is attained. Therefore, the drag load due to spin-up or spring-back combines with earlier lower-magnitude vertical loads but does not combine with the maximum vertical load.

The tail-down landing situations of the Basic Landing Conditions table shown in Appendix C of Part 23 appear to erroneously justify the first interpretation (above) because the lack of a load for D_r . KnW = 0 differs from Wheel Spin-Up and Spring-Back Loads, Appendix D of Part 23, which offers non-zero drag loads for the same landing conditions shown in the Appendix C figures of Part 23.

When a landing airplane wheel touches the ground, a near instantaneous wheel spin-up occurs (since the wheel tangential velocity should quickly match that of the runway). Due to inertial properties, this phenomenon induces a drag load upon the landing gear. The drag force energy is stored in the landing gear as potential energy that causes the gear to spring-back (a negative drag force). The drag forces of spin-up and spring-back may reach maximum values at different times than when the vertical load on the landing gear is achieved. It would be unusual for both load maximums (drag and vertical) to occur simultaneously. Airplane landing gear designers often estimate these loads analytically and test for these loads by dropping the airplane or the landing gear units. The drag loads are sometimes approximated using inclined planes (wedge blocks) and, at other times, they are created by reverse spinning the wheel and tire before the drop impact.

Incline planes are often suitable for small airplanes where the touchdown velocity is relatively small and the wheel and tire diameters (and thus inertial forces) are correspondingly small. Incline planes, which are used to induce the estimated drag load, restrict proper development of the spring-back (negative drag) load—and should be viewed accordingly. For small airplane landing gear designs, incline planes may be an adequate method to use to show compliance with Part 23.

Drop tests using reverse wheel spin-up are the most accurate way to simulate the behavior of the landing gear and the combined vertical and drag loads caused by wheel spin-up and spring-back. This method is frequently used by landing gear designers for larger, faster, airplane designs. Higher touchdown speeds and larger diameter tires cause increases in the wheel and tire inertial properties and create larger magnitude spin-up and spring-back loads.

> **Policy:** Reverse wheel spin-up and incline planes are both acceptable methods for imposing the drag inputs into a drop test when they are properly applied. Also, see § 23.725(c).

23.483 One-wheel landing conditions (Original)

Use wing lift consistent with that used for the other landing impact conditions.

23.485 Side load conditions (Amendment 23-45)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

(d) No policy available as of June 1, 1994.

23.493 Braked roll conditions (Original)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

23.497 Supplementary conditions for tail wheels (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

23.499 Supplementary conditions for nose wheels (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

(d) No policy available as of June 1, 1994.

(e) No policy available as of June 1, 1994.

23.505 Supplementary conditions for skiplanes (Amendment 23-7)

A ski installation should use a load factor determined by either of the following two methods:

- Perform drop tests, with skis installed, on a surface simulating frozen hard-packed snow or ice; or
- Use a conservative formula (see SPECIFICATION—Aircraft skis, National Aircraft Standards Committee, NAS 808, paragraph 5.1(a)).

Note: Ski installation factors should include consideration for fittings, tubes, axles, nuts, bolts, etcetera, which attach the skis to the fuselage. Ski-gear loads normally run about 115 percent to 125 percent of wheel-gear loads.

Also, see 23.737, Skis.

23.507 Jacking loads (Amendment 23-14)

(a) No policy available as of June 1, 1994.

(b) The intent is accomplished when structure affected by jacking loads is designed to withstand the inertial load factors in 14 CFR Part 23, § 23.507(a)(2) and (c).

(c) No policy available as of June 1, 1994.

23.509 Towing loads (Amendment 23-14)

(a) No policy available as of June 1, 1994.

(b) Auxiliary gear means a landing gear unit that is not part of the main landing gear, that is, a tail bumper on a tricycle gear airplane or wing protectors on a glider.

(c) No policy available as of June 1, 1994.

(d) No policy available as of June 1, 1994.

23.511 Ground load; unsymmetrical loads on multiple-wheel units (Amendment 23-7)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

WATER LOADS

23.521 Water load conditions (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

See 23.603, Materials and workmanship, for information that pertains to floats.

23.523 Design weights and center of gravity positions (Amendment 23-45)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

23.525 Application of loads (Amendment 23-45)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

(d) No policy available as of June 1, 1994.

23.527 Hull and main float load factors (Amendment 23-45)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

23.529 Hull and main float landing conditions (Amendment 23-45)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

23.531 Hull and main float takeoff condition (Amendment 23-45)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

23.533 Hull and main float bottom pressures (Amendment 23-45)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

23.535 Auxiliary float loads (Amendment 23-45)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

(d) No policy available as of June 1, 1994.

(e) No policy available as of June 1, 1994.

(f) No policy available as of June 1, 1994.

(g) No policy available as of June 1, 1994.

23.537 Seawing loads (Amendment 23-45)

No policy available as of June 1, 1994.

EMERGENCY LANDING CONDITIONS

23.561 General (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) Guidance about providing every reasonable chance of escaping serious injury.

Cargo Restraints

Guidance information about ultimate inertial strength requirements for cargo restraint devices in 14 CFR Part 23 airplane designs originally centered around 14 CFR Part 135 air taxi operators who carried mail sacks in an empty cabin after passenger seats were removed. The guidance is (around 1968):

Discussion: There is a 4.5g inconsistency between the 9g occupant protection provisions from mass items in the cabin and the 4.5g cargo compartment items in Civil Air Regulations (CAR) 3.386(d) and CAR 3.392. This inconsistency carried to § 23.561(e) and § 23.787(c) in recodified Part 23, dated February 1, 1965.

As originally viewed, CAR 3.392 applied to airplane configurations that contained a forward crew compartment, a center passenger compartment, and an aft bulkhead that separated a small cargo compartment from the passenger area. In this configuration, 4.5g restraint was considered adequate since the National Aeronautics and Space Administration (NASA) data showed that, in a typical crash, the g-forces became lower as distance increased aft from the airplane nose. Thus, what would be adequate cargo restraint in the aft fuselage would probably be inadequate near the cockpit. When CAR 3.392 was adopted, the Civil Aviation Authority did not envision all-cargo CAR Part 3 aircraft.

Guidance: For § 23.561(b) and (e), the restraining devices should meet the 9g requirement. The up (3g) and side (1.5g) load factors given in § 23.561(b)(2) only apply if crewmembers would be subjected to potential injury from upward or sideward cargo movement.

Advice: Remember to consider the crew emergency exits that are required by § 23.807(a)(3).

Cargo Compartment Design

Regulatory history about cargo compartment design practices and minimum airworthiness standards (around 1968):

Discussion: When structure separates the contents in a cargo compartment from occupants forward of the compartment, CAR 3.392 requires a cargo restraint of 4.5g forward inertial load factor and § 23.787 requires a cargo restraint of 9.0g forward inertial load factor. At Amendment 23-36, the FAA increased the mass item retention to 18.0g for occupant protection when the mass items are located in the cabin with the occupant (that is, not located in a separated cargo compartment).

History: Civil Aeronautics Manual (CAM) 3.392-1 referred to the variation in load factors between emergency crash conditions and cargo compartment requirements. It stated that CAR 3.392 was specially promulgated to overcome objections to the excessively heavy cargo compartment structure required to meet 9g's. Further, passenger injuries were not prevalent with the lower design factors of Aeronautical Bulletin 7a, dated July 21, 1929, and CAR Part 4a.

FAA experience indicates that many of the aircraft designed to Aeronautical Bulletin 7a or CAR 4a were somewhat overstrength. This resulted from the state-of-the-art design practices, coupled with a design philosophy of robust structure during that period. Rather than establishing the adequacy of the lower load factor for such compartments, it may be that the lack of injuries noted by CAM 3.392-1 merely highlight the overstrength cargo compartments in the older airplanes.

While the seats and all other mass items in the fuselage should be good for 9g's, it is difficult to argue that the cargo in the rear of a passenger compartment need only be restrained for a 4.5g load factor.

Emergency provisions protection—the intent of CAR 3.386 is as follows: Each acceleration specified in CAR 3.386 should be considered to act independently. If an unsafe feature exists because the occupants' heads could strike sharp edges of objects within the passenger cabin under combinations of load factors less than those given in CAR 3.386(a), then the designer should use the general requirement of the first paragraph of CAR 3.386 with the guidelines of CAM 3.386-1.

Cabin Safety

Airplane designers (and modifiers) and FAA Aviation Safety Engineers should pay attention to the need for increased emphasis about the design of cabin safety provisions in small airplanes.

Observe the following guidelines

Strive for the highest level of occupant crash protection feasible within the state-of-the-art technology with respect to the general emergency landing conditions and the occupant protection provisions given in §§ 23.561 and 23.785. Do this early and frequently in the Type Certification Process.

Cabin safety emphasis is intended to influence an individual designer's choices early in the conceptual phases, when safety objectives can be achieved with little or no burden on the designer and the manufacturer. The level of occupant crash protection is determined by interrelated cabin features including, but not limited to, the following:

- Seating configuration (considering occupant flailing characteristics);
- Occupant restraint devices, supports, attachments, and installations;
- Energy absorbing padding (thoughtfully located); and
- Potentially hazardous hard points.

Qualitative judgment should be exercised while one considers the advantages of occupant dynamic response analyses, the use of human tolerance data, and the energy absorption properties of padding and structure. In a transport airplane certification program, the applicant found it feasible to substantiate compliance with occupant head protection requirements by a combined use of model testing and human tolerance data. This substantiation considered state-of-the-art technology and safety objectives that are applicable to general aviation airplanes.

A Summary of Crashworthiness Information for Small Airplanes, published by the FAA, which contains information that designers can use to improve cabin safety, appears in Figure 9.

Clarify confusion about static side load factors: Mass items have a 4.5g static side load factor requirement, whereas occupants have only a 1.5g condition.

The current occupant sideward static load factor requirement of 1.5g was recodified into § 23.561(b)(2)(iii) from the Civil Air Regulations, Part 3.

Part 23, Amendment 23-36, added the 4.5g requirement to protect the occupants from items of mass within the cabin that could injure the occupant. The General Aviation Safety Panel (GASP) studied and recommended these actions. GASP I, as the first group was known, recognized that 26g times the Sin 10° equals 4.5g, the sideward dynamic emergency landing condition for an airplane impact associated with a yaw. They further recognized that it would be much more economical to conduct static tests, or analyses, than dynamic tests for mass items installed on the airplane. Consequently, GASP I recommended the static load factor now contained in § 23.561(b)(3).

(c) No policy available as of June 1, 1994.

(d) Protection of occupants in an airplane with a sliding bubble canopy and with no turnover structure principally involves the following:

Background: Only a few low-wing airplanes with tricycle landing gear, sliding canopies, and no turnover structure, have been certificated in the U.S. Until

Amendment 23-36, effective September 14, 1988, the turnover requirement in § 23.561(d) was essentially the same as in CAR 3.386(c), which first appeared in CAR 3, as amended in November 1949.

NOTICE **DEPARTMENT OF TRANSPORTATION**
FEDERAL AVIATION ADMINISTRATION

210A file

N 8130.21

6/7/73

Cancellation
Date: 12/1/73

SUBJ: CABIN SAFETY FOR SMALL AIRPLANES

1. PURPOSE. This notice calls attention to the need for increased emphasis on the design of cabin safety provisions for small airplanes.

2. DISTRIBUTION. This notice is distributed to the branch level and above in the Headquarters Flight Standards Service; Western Region Aircraft Engineering Division; European Region Aircraft Certification Staff; and to section level and above in regional and center Flight Standards offices.

3. ACTION

a. Personnel responsible for findings of airworthiness compliance, including industry personnel with delegated authority, should make every effort within the framework of existing certification procedures and objectives defined in FARs 23.561 and 23.785 to secure for airplanes undergoing type certification a high level of occupant crash protection feasible within the state of the art. Arrangements should be made to assure for each type certification program that FARs 23.561 and 23.785 are accorded special attention throughout the program. This should be treated thoroughly early in the program, especially during the Preliminary Type Certification Board Meeting, in order that the applicant may have sufficient lead time to fully achieve regulatory safety objectives.

b. This emphasis on cabin safety is not intended to be defacto upgrading of existing regulations, but rather, a shift of emphasis in order to preclude the approval of hardware which is marginal for the crash condition. It should emphasize choices which with little burden on the manufacturer could more assuredly meet safety objectives through prudent choice of feasible design alternatives. The level of occupant crash protection for an airplane cabin is determined by numerous interrelated cabin features, principal of which are seating configuration (as related to flailing characteristics of occupants), installation features of occupant restraint devices and supports, and the location of energy absorbing padding and potentially hazardous hardpoints. While assessment of these features depends to a large degree on the qualitative judgment of the specialist responsible for finding compliance, this assessment can be refined and qualified through analysis of occupant dynamic response and the use of data on human tolerance and energy

Distribution: WFS-3, AWE-100 (10 copies); AEU-100 (10 copies); RCFS-4; AAC-955 (80 copies) Initiated By: AFS-120

GPO 887-904

FIGURE 9 (Page 1 of 2)

N 8130.21 6/7/73

absorption properties of padding and structure. In a recent transport certification program, the applicant found it feasible to substantiate compliance with occupant head protection requirements by a combined use of model testing and human tolerance data. Although a transport configuration was involved, this substantiation considered safety objectives and a state of the art which are identical to those applicable to general aviation airplanes.

c. This notice does not imply that any particular substantiation method should be required for aircraft certification. It does, however, emphasize that protection as envisaged in FARs 23.561 and 23.785 is of utmost importance and that every effort should be made to utilize the latest state of the art to achieve a high level of occupant protection under these regulations.

4. BACKGROUND

a. During the recent growth of aviation and introduction into service of the new generation airplanes, considerable work by FAA and industry was directed toward achieving a higher level of cabin safety for the increasing numbers of passengers with emphasis on crash protection features for transports affecting the widest cross section of the general flying public. It is apparent that the smaller general aviation airplanes should be accorded an increased amount of attention in the crashworthiness program and in particular that special attention should be given during type certification to occupant crash impact protection. Crash protection standards have been included for some time in CAR 3.386 and in FARs 23.561 and 23.785, as objective standards to be applied to the extent necessary for a high level of safety commensurate with the state of the art. Advances in technology in recent years have made feasible higher levels of crash protection, and these higher levels should be included in the newer airplanes. However, in some cases they are not, and the effectiveness of the cabin safety program has been questioned in this regard.

b. To improve the cabin safety program from an overall standpoint, a publication by the FAA, "A Summary of Crashworthiness Information for Small Airplanes," contains considerable information of use to designers.

C. R. Melugin Jr.

Acting Director, Flight Standards Service

FIGURE 9 (Page 2 of 2)

In the past, FAA Certifying Offices have generally accepted that turnover is not reasonably probable. Most U.S. designs contain some cabin structure and a sideward opening door, which has been a relieving consideration when applying these regulations.

High-wing tricycle gear airplanes will turnover. Accident and incident reports indicate that this occurs fairly regularly. However, small high-wing airplanes usually have adequate wing structure and strength to meet the emergency landing conditions of § 23.561(d) and protect the occupants in a complete turnover.

Figure 10 presents an overall relational view of turnover protection airworthiness standards.

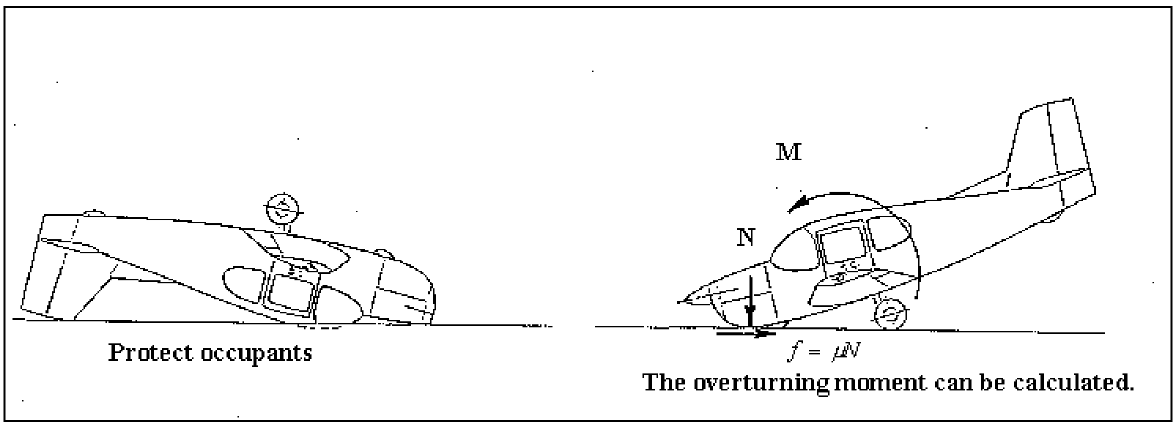

FIGURE 10

The FAA emphasizes protecting the occupants. Less stringent attention is paid to solving the problem of the airplane overturning, which is more complicated in 3-dimensions than the simplified 2-dimensional sketch shown.

Regardless of the kind of landing gear installed, the inverted attitude is probable; therefore, the emergency exit requirements of § 23.807(a) should be demonstrated unless escape means are obvious.

Discussion: The FAA Accident/Incident Data System (AIDS) provides the following 5-year history, ending February 10, 1984, about nose-up or nose-over occurrences. Classification ~ Nose-Up-Over:

Low-Wing, Tricycle Landing Gear (no amateur-built, agriculture airplanes or illegal operations)

Nose-over	56
Nose-up	6
Undefined	61
Total cases	123
Injuries	77
Fatalities	1
Airport landing or takeoff (includes under or overshoot)	58
Emergency landing (and off-airport operations)	45
Undefined	11

High-wing, turnover: The data for a high-wing tricycle landing gear, two-place trainer series, shows 217 turnover occurrences with 31 injuries and no fatalities. The data for high-wing airplanes with a conventional (tail wheel) landing gear shows 239 turnovers with 16 injuries and no fatalities.

Conclusion: Note that the injury frequency for high-wing airplane accidents is considerably lower than that for low-wing airplane accidents. The total high-wing airplane accident experience (for this study) is about 10.3 percent injuries per turnover (47/456, with no fatal injuries) versus a low-wing accident experience of about 63 percent injuries per turnover (78/123, including one fatal injury).

National Transportation Safety Board (NTSB) safety recommendations A-81-26 through A-81-28 reported emergency egress difficulties. In four of six accidents, the airplanes came to rest inverted. In these accidents, the airplanes struck obstacles during the approach or initial climb flight phase.

Policy: The airplane structure should be designed to protect the occupants during any survivable accident, regardless of the flight phase.

To meet the static emergency landing conditions requirements (see § 23.561(d)), the strength of turnover structure may be shown by analysis. However, conduct tests if an analysis is questionable (see § 23.601).

Comments about a Jettisonable Canopy

The airworthiness standard for § 23.807(a), Emergency exits, requires that emergency exits should be located to allow escape in any probable crash attitude. Compliance with this requirement should be demonstrated with a rollover structure installed.

Section 23.807(b)(4), the emergency exits regulation, requires each emergency exit to have reasonable provisions against jamming by fuselage deformation. Compliance with this requirement should be demonstrated with the airplane resting inverted after accounting for any structural collapse.

In an acrobatic category airplane, the occupants should be able to bail out quickly, with parachutes, at any speed between V_{SO} and V_D (see § 23.807(b)(5)). Further, § 23.807(c) requires that the proper functioning of each emergency exit be shown by tests, which are usually demonstrated on the ground rather than in flight.

Recommendation: Forward sliding, jettisoning, and hinged canopy designs were not originally envisioned by CAR 3 or Part 23. Occupant protection and escape from an airplane damaged in a turnover, and in-flight emergency escape provisions should be addressed in any certification project, since adequate emergency exit airworthiness standards exist in these regulations. Special conditions may be required for a jettisonable canopy to address continued safe flight, inadvertent canopy opening hazards, and in-flight canopy jettison-safe trajectory. Other possible methods of showing compliance to the airworthiness standards include wind tunnel tests, sled tests, or other ground tests that simulate flight.

A jettisonable canopy may not jam when the airplane is resting inverted unless there are other provisions for egress.

With respect to the bail-out requirement of § 23.807(b)(5) for an acrobatic category airplane:

- Show that the occupants can safely exit the airplane between V_{SO} and V_D.

- If the canopy is jettisonable, show that the canopy trajectory will not injure the occupants while separating from the airplane between V_{SO} and V_D. Also, demonstrate that the airplane can continue safe flight and landing without the canopy. Alternatively, inadvertent canopy jettison should be improbable.

Emergency Exit Requirements in an Airplane Turnover Condition

Guidance: Regardless of the type of installed landing gear, the inverted attitude is a probable crash attitude for small airplanes. More importantly, the occupant emergency exit requirements of § 23.807(a) should be shown. If escape from the inverted attitude airplane is not obvious, or is questionable, compliance should be by demonstration.

It is not acceptable to rely on an emergency procedure that requires canopy jettison immediately before an impact accident (except for an in-flight canopy jettison above a safe parachute altitude required in § 23.807(b)(5)). If the canopy is made jettisonable, to comply with § 23.807(b)(4), avoid jams with fuselage deformation when the airplane is resting inverted. If there is any doubt, the applicant should demonstrate by tests.

(e) No policy available as of June 1, 1994.

Do not allow an internal cabin door to jam during an emergency landing and block the flight crew's escape path. See the Systems and Equipment Guide for Certification of Part 23 Airplanes, AC 23-XX-29, 23.807 Emergency Exits for further guidance.

Policy: With respect to 170-pound and 190-pound occupants:

- Use, at least, the following minimum occupant weights when showing compliance with the emergency landing conditions static strength airworthiness standards of § 23.561:

- Design each seat and its supporting structure for an occupant weight of at least 170 pounds, for normal and commuter category airplanes. Use a 190-pound occupant weight, which includes a parachute, for utility and acrobatic category airplanes. (Reference § 23.25(a)(2).)

- Also, design each seat and restraint system for these occupant weights when considering maximum flight and ground-load conditions of the airplane-operating envelope. **Note:** A 1.33 factor should be applied to all loads that affect the strength of fittings and attachment of the following:

  (1) Each seat to the structure, and

  (2) Each safety belt and shoulder harness to the seat or structure.

23.562 Emergency landing dynamic conditions (Amendment 23-50)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

(d) No policy available as of June 1, 1994.

(e) Acceptable Alternate Methods

Certification of energy absorbing seats and restraint systems began in 1992. Currently, the FAA is not accepting compliance with this requirement by analysis alone. At this stage of limited experience, the FAA sees analytical models as useful for seat and restraint system design, development, determining critical test conditions, and reducing development costs by the minimizing number of test attempts; however, tests are required for certification. Future certification compliance by analysis alone will be dependent on the accumulation of a sufficiently large database that has been well correlated with test results. It will also depend on other factors, such as the experience of the analytical model user.

FAA sponsored the development of computer analysis programs called Seat/Occupant Model-Light Aircraft (SOMLA) and Seat/Occupant Model-Transport Aircraft (SOMTA). A copy of a paper titled, "Analysis of Aircraft Seats and Restraint Systems Using Programs SOMLA/SOMTA," by David H. Laananen, Department of Mechanical and Aerospace Engineering, Arizona State University, is available. It may be obtained by mail from the following: Manager, Standards Office (ACE-110), Federal Aviation Administration, Small Airplane Directorate, DOT Building, 901 Locust, Room 301, Kansas City, MO 64106.

The following two SAE papers provide helpful, related, information:

- *SAE Paper No. 850853*, "The Development of Dynamic Performance Standards for General Aviation Aircraft Seats," Stephen J. Soltis and John W. Olcott.

- *SAE Paper No. 851847*, "Human Injury Criteria Relative to Civil Aircraft Seat and Restraint System," Richard F. Chandler.

AC 23.562-1, Dynamic Testing of Part 23 Airplane Seat/Restraint Systems and Occupant Protection, contains useful guidance about this airworthiness standard topic.

Alternate Means of Compliance—Restricted Category Airplane

The FAA issues type certificates under the regulatory procedures of 14 CFR Part 21, Certification Procedures for Products and Parts.

The Small Airplane Directorate decides the appropriateness of 14 CFR Part 23 airworthiness standards as they apply to the special purpose restricted category airplane. If a regulation is found inappropriate, the FAA excludes that regulation from the certification basis of that airplane. Note: A finding of equivalent level of safety or the need for an exemption is unnecessary for that specific airworthiness standard.

Special Purpose Agricultural Airplanes

The FAA has allowed the certification of restricted category aircraft since Civil Air Regulations (CAR) Part 8 was introduced (around 1950). The preamble to CAR 8 recognized that, for restricted category aircraft where the public was not endangered, it was unnecessary to provide an equivalent level of safety to the standard airworthiness requirements.

Policy: As of December 1, 1997, for emergency landing dynamic conditions, evaluate the airplane with at least the following considerations:

(1) The placement of the chemical hopper forward of the cockpit so that there is no large item of mass that threatens to collapse the cockpit should a crash occur.

(2) The elimination of protruding knobs, handles, or other rigid structures in the cockpit with which the pilot or crew member may come into contact, in the event of a crash. Approved Department of Transportation or Mil-Spec protective headgear is mandatory.

(3) Installation of a military type lap belt and shoulder harness having a 5,000-pound rating or greater or approved equivalent.

(4) Special purpose crew members who assist in the aerial application operation, that is, flaggers, loaders, can be carried in ferry flights provided that each crew member has a seat, a lap belt and shoulder harness comparable in strength to that of the pilots, the crew seat is not in the cockpit, and the crew seat is located behind the pilot seat. Special purpose crew members that are carried for any other purpose will be afforded the same protection as that of the pilot.

Regarding agricultural airplane designs, the Small Airplane Directorate decides whether or not to exempt § 23.562, Emergency landing dynamic conditions, from a specific make and model airplane.

Dynamic seat airworthiness standards used in Part 23 were developed for normal, utility, and acrobatic category airplanes only. These standards were never intended for use on restricted category airplanes. When NASA, the general aviation industry, NTSB, and the FAA examined survivability envelopes, they did not consider restricted category airplanes in the database because there were notable differences in the crash scenarios.

The Small Airplane Directorate reaches the conclusion to exempt or not exempt after reviewing crashworthiness design features of the specific make and model airplane. Service experience shows that agricultural airplane operators have a lower accident fatality rate than general aviation operators. Certain make and model agricultural airplane designs have contained increasingly effective crashworthy features throughout recent development history, which are the obvious reasons for the fatality rate differences.

Policy: The FAA will not automatically exempt the emergency landing dynamic conditions requirement (§ 23.562) for any applicant who is seeking a type certificate for an agricultural airplane. Instead, the Small Airplane Directorate reviews the design for compensating features to the dynamic seat airworthiness standards and decides to exempt or not exempt on a case-by-case basis.

For useful guidance, refer to AC 21.25-1, Issuance of Type Certificate: Restricted Category Agricultural Airplanes.

FATIGUE EVALUATION

23.571 Metallic pressurized cabin structures (Amendment 23-48)

(a) Fatigue Evaluation.

History: Fatigue evaluation of pressurized cabins was first required for small airplanes by Amendment 3-2 of the Civil Air Regulations (CAR), Part 3, effective August 12, 1957, and it continued into the original Part 23.

Safe-life requirements mandate that certain critical structural elements have a fatigue life determined during the airplane type certification process. Life-limited items are normally identified in the Type Certificate Data Sheet Note 3. As of Amendment 23-26, 14 CFR Part 23, § 23.1529 requires time-limited items to be shown in the Instructions for Continued Airworthiness—Airworthiness Limitations Section.

Policy: The FAA allows an extension of the originally imposed safe-life limits only through a formal reinvestigation of the life-limited parts. Generally, the type certificate holder conducts a new fatigue certification program. Replacing the safe-life structural elements is a viable alternative to recertification. However, this is not always the best course of action because some major structural elements, like the wing and fuselage pressure vessel assemblies, are not easily or economically replaceable.

(b) Same as in (a).

(c) Amendment 23-45, effective September 7, 1993, provides this airworthiness standard as an option to § 23.573(b), Damage tolerance and fatigue evaluation of structure.

AC 23-13, Fatigue and Fail-Safe Evaluation of Flight Structure and Pressurized Cabin for Part 23 Airplanes, contains useful guidance about this airworthiness standard topic.

23.572 Metallic wing, empennage, and associated structures (Amendment 23-48)

(a) Fatigue Strength

History:

Civil Air Regulations (CAR) 3, Amendment 3-2, effective August 12, 1957, first imposed a fatigue evaluation of pressurized cabin airplane designs.

14 CFR Part 23 adopted an airworthiness standard to evaluate fatigue of an airplane wing and associated structure at Amendment 23-7, effective September 14, 1969.

Amendment 23-34, effective February 17, 1987, added commuter category airplanes to Part 23. Empennage fatigue requirements were included for these airplanes. SFAR 41, which applied to Part 23 derivative-model airplanes, always had such a requirement. Effective October 26, 1989, the FAA issued Amendment 23-38, which extended the fatigue requirement for empennage, canard surfaces, tandem wing, winglets, and tip fins to all Part 23 airplanes. Amendment 23-45, effective September 7, 1993, added the option of a damage tolerance evaluation, as defined in § 23.573(b). Amendment 23-48, effective March 11, 1996, made damage tolerance evaluation mandatory for commuter category airplanes.

Narrative about spar-component fatigue tests combined with the safe-life and fail-safe design philosophies displays the FAA intentions of the time period before 1971.

> Component testing, while acceptable under certain conditions, presents the problem of determining which structure to test and how to ensure that the correct testing conditions are applied. The FAA intends for the airplane designer to show that the wing, wing carry-through structure, and attaching structures comply with the fatigue requirements (§ 23.572). These examinations may exclude the control surfaces and their attachments.
>
> These examinations normally include the main spar, the secondary spar, stringers, torque box skin, and at least the main internal ribs. While a main spar-component test could adequately substantiate the spar, the remaining structures should be proven by additional component tests, or analyzed as either safe-life or fail-safe structures. If fail-safe compliance is chosen, determine if the specified loads can be supported with a failed element. Also, determine the number and kind of inspections to find the damage before catastrophic failure.

When testing the main spar, simulate load transfer through skin attachment units and the associated fretting. Further, consider and simulate any significant eccentricities and rib-to-spar-cap loads.

Exercise good judgment to ensure that elements and aspects of primary importance to safety receive the most emphasis.

Safe-Life Limits—Two Lives

When a safe-life limit is established for airplane designs certified in both normal and utility categories, show the lower of the two lives in the TCDS and show the following note in the TCDS, the Airplane Flight Manual, and the Airworthiness Limitations Section of the Instructions for Continued Airworthiness (when they exist): "**Since the airplane is type certificated under both normal and utility categories, the lower fatigue life has been listed in the TCDS.**"

Wood Structure and Fatigue

Research indicates wood is not sensitive to fatigue if the stress levels are low. Emphasis should be placed on ultimate load tests and environmental protection of the wood to prevent deterioration due to dry rot and other related environmental factors.

Guidance About Tail Fatigue in Modified Small Single-and-Twin-Engine Airplanes with Higher Horsepower Engines

A gross weight increase or an increase in airplane speed frequently demands an increase in horsepower. An engine horsepower increase may significantly relocate the engine center of gravity or increase engine weight with respect to the original installation, or both. A different engine mount (stiffness change) may accompany a different engine installation. When engine c.g., engine weight, or the engine mount are changed, one should consider the benefits of a pre-modification and post-modification ground vibration survey. The purpose of the survey is to identify any coupling or resonant characteristics changed by a different engine installation (that is, nodes, modes, and frequencies).

The effects of fatigue upon airplane empennage structure, due to propeller slip-stream impingement, could be assessed by an in-flight vibration monitoring program or a flight-strain survey. These in-flight tests could also give insights about whether increased horsepower forced vibrations affect airplane critical vibration environments that were previously benign.

One should also compare the in-flight torsional and bending peak stresses, and some selected panel strain-gauge readings with the original airplane design data. The objective is to verify that the new stress levels will not adversely affect the fatigue life of the empennage.

Often, a modifier neither has, nor can they get, access to the original airplane design data. Consequently, other means of showing compliance to the airworthiness standards may be used. One evaluation technique is to compare the new fatigue stresses to the material endurance level, that is, the S-N curves. A ground vibration survey and a flight-strain survey conducted before and after the modification can provide some data to perform a comparative analysis. If changes in the stresses are large enough to affect the empennage fatigue life, the modifier should determine appropriate structural design changes to include in the modification. It is here that good engineering judgment should be exercised. The service history of a similar type airplane incorporating a like modification may be used to identify potential fatigue crack locations and to serve as a guide when preparing detailed structure inspection methods and frequencies.

Fatigue critical structure is defined as structure whose failure would cause catastrophic loss of the whole airplane. Critical structure would include the spar, the primary fittings, the pressurized fuselage skin-stringer combinations, and the frames.

Interpretation

Instead of a fatigue or fail-safe strength investigation, § 23.572(a) permits compliance by showing that the structure, operating stress level, materials, and expected use are comparable, from a fatigue standpoint, to a similar design that had extensive satisfactory service experience.

FAA Report No. ACE-100-01, “Fatigue Evaluation of Empennage, Forward Wing, and Winglets/Tip Fins,” contains comprehensive guidance on this subject. It is available from the National Technical Information Service, Springfield, VA 22161.

See § 23.571(a), Metallic pressurized cabin structures, for FAA guidance about safe-life airplane fatigue limitations imposed during the type certification process of civil aircraft.

Fail-safe strength: No policy available as of June 1, 1994.

Damage tolerance:

Effective September 7, 1993, Amendment 23-45 provided the damage tolerance option in § 23.573(b).

(b) No policy available as of June 1, 1994.

AC 23-13, Fatigue and Fail-Safe Evaluation of Flight Structure and Pressurized Cabin for Part 23 Airplanes, contains comprehensive guidance about this airworthiness standard topic. Please refer to it for additional information. **Until AC 23-13 is revised to convey the following information, it is presented here as a**

courtesy: Report AFS-120-73-2, "Fatigue Evaluation of Wing and Associated Structure on Small Airplanes," contains fatigue load spectra for various Part 23 airplane usage categories. It also contains detailed procedures for the fatigue strength (safe life) investigation of § 23.572(a)(1). A computer program that performs the report calculations is available from the following: Manager, Standards Office (ACE-110), Federal Aviation Administration, Small Airplane Directorate, DOT Building, 901 Locust, Room 301, Kansas City, MO 64106.

Airplane maintenance record entries:

Should a pilot make airplane maintenance record entries about airplane operations that relate to the established life-limits of the airplane according to instructions in a Pilot Operating Handbook?

A type certificate applicant should show compliance with § 23.572 (fatigue life limits) using certification procedures for airplane design. The FAA does not allow a type certificate applicant to impose a requirement for the pilot to record aerobatic flight time in the airplane maintenance record. It is not possible for the FAA to validate these kinds of airplane record entries. Maintenance records, in the 1997 version of Part 91, require the following registered owner or operator entries for the airframe, each engine, each propeller, and each rotor:

(1) Total time in service; and

(2) Current status of life-limited parts.

23.573 Damage tolerance and fatigue evaluation of structure (Amendment 23-48)

(a) Since Amendment 23-45 (effective September 7, 1993), the damage tolerance and fatigue evaluation of composite structure has been based on the applicable requirements of 14 CFR Part 23, § 23.573. AC 20-107A, Composite Aircraft Structure, which existed before Amendment 23-45, contains acceptable means of showing compliance with these requirements. One consideration that is different for composite materials than for metallic structures is that of impact damage. For composite structures, impact damage resulting from events such as dropped tools or hail impacts is difficult to detect, but may cause degradation of static or fatigue strength. Another difference for composite structures is that, in addition to tensile loads, compressive loads may also drive damage growth. When demonstrating compliance with the growth rate or no-growth rate of damage requirement (§ 23.573(a)(2)), it is important to consider compressive loads that may drive the growth of disbonds or delaminations in composite structures.

With respect to movable control surfaces, include any structure that, if it failed, would cause loss of the airplane.

(b) There is no guidance for metallic structures damage tolerance assessment for Part 23 airplanes. Presently, a Transport category airplane (14 CFR Part 25), AC 25.571-1C, Damage-Tolerance and Fatigue Evaluation of Structure, dated April 29, 1998, provides the only information available from the FAA.

23.574 Metallic damage tolerance and fatigue evaluation of commuter category airplanes (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

23.575 Inspections and other procedures (Amendment 23-48)

No policy available as of June 1, 1994.

SUBPART D—DESIGN AND CONSTRUCTION

23.601 General (Original)

The manner of testing can bias the test results. Several years ago, the wings failed on an FAA-approved airplane whose pilot was performing aerobatic maneuvers with a full load of passengers. When comparing the static test results with the pattern of failure of the wing, investigators suspected that the tension pads used to apply wing-bending loads likely stabilized the upper wing skin during tests. The stabilizing effect kept the upper wing skin from buckling during the tests. Additional static tests verified the foregoing theory; the retested wing failed below ultimate load.

When approving any static tests set up for thin-skinned structure, consider the effects of installing tension pads because they may contribute a stabilizing effect upon the structure and bias the test results.

See 23.307, Proof of structure, for secondary structure guidance.

23.603 Materials and workmanship (Amendment 23-23)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

See AC 20-44, Glass Fiber Fabric for Aircraft Covering, for additional guidance about this airworthiness standard topic.

Floats

Both Technical Standard Order TSO-C27 and National Aircraft Standards NAS 807, “Twin Seaplane Floats,” and AC 20-107A, Composite Aircraft Structure, contain guidance and criteria for floats constructed from composite materials. While NAS 807 does not specifically address composite materials, the standards in Section 3, Material and Workmanship, and in Section 4.2, Strength, are general enough to apply to materials other than the conventional aluminum materials normally employed.

Composite materials require special considerations for handling and storage that are not commonly required for metallic materials. These factors may affect material and process specifications.

In addition to the 1.5 factor of safety, another factor could be applied for material variability when substantiated by tests. Environmental effects include moisture, saltwater exposure, ultraviolet light, temperature variations, material composition, and geometric dimensions. These factors can be required under Section 4.2.1, Material Strength Properties, and Section 4.2.3.1, Special Factors, of NAS 807.

Component static strength tests should include defects such as debonds and voids. They should also reflect impact damage, up to the threshold of detectability, for the inspection system in use during manufacturing and operations. Both defects and impact damage should be located in critical areas that are expected as a result of production assembly bonding processes, and in operational service conditions. The designer should identify the nature and size of such defects, and damage. The manufacturer should have an inspection system functioning during production and operational service. This system should ensure that defects do not reduce structural strength below ultimate load. Critical areas should include bonding of support strut attachment fittings to the basic structure. An alternative approach would be the use of mechanical fasteners, in critical areas, where bonding defects would cause critical loss of strength.

Galvanic corrosion may occur when unprotected metal is in contact with graphite composite material in a corrosive environment. Affected metal parts will need suitable protection. Cadmium plated metals will corrode. Use fasteners made of corrosion resistant materials (for example, titanium or corrosion resistant steel).

Also, the bonds of aluminum parts in contact with composite material (including fiberglass) may seriously degrade over time due to moisture absorption. Special treatment (for example, phosphoric anodizing) of the aluminum is necessary to maintain the original strength of the bond.

Fatigue requirements appearing in NAS 807, Section 4.2.1, are minimal; they are also identical to the Civil Air Regulations (CAR) 3.307 general requirement for airframe construction. The FAA has generally not required fatigue testing or analysis to meet this requirement. Considering that float service history has generally been satisfactory and that more stringent fatigue requirements have not been applied to Part 23 airplane landing gear, AC 20-107A, Section 6, Proof of Structure - Fatigue, does not apply. However, the manufacturer should develop instructions for appropriate tests or inspections to detect problems of hidden damage or delaminations. Then the manufacturer should also develop repair instructions for the composite material structures. For float design, maintaining the integrity of watertight compartments is a special concern--debonding or delaminations that cause inter-compartment leakage may reduce the level of safety intended by 14 CFR Part 23, § 23.751 and CAR 3.371.

There are no requirements for maintenance instructions under the TSO general requirements (refer to 14 CFR Part 21, Subpart O) or in both TSO-C27 and NAS 807 specifications for seaplane floats. However, § 21.50(b) requires that Instructions for Continued Airworthiness be prepared by the applicant according to § 23.1529, and that they be provided with each supplemental type certificate (STC) applied for after January 28, 1981.

See AC 20-107A for additional guidance about this airworthiness standard topic.

Policy: All composite structures that are critical to flight safety should be designed to be damage tolerant.

If impractical, the applicant is referred to § 23.573(a)(6), Damage tolerance and fatigue evaluation of structure. The manufacturer should substantiate scatter factors.

Consider the following items when demonstrating the damage tolerance capability of structures critical to safe flight:

Introduce manufacturing defects and realistic impact damage up to the threshold of detectability.

Substantiate ultimate load retention capability after one demonstrated lifetime of in-service usage.

Introduce initially detectable damage.

Apply a statistically significant number of flight-by-flight spectrum repeated-load cycles to validate one lifetime or an inspection interval for operational service use. The number of cycles applied should be determined by either load or life considerations. If the damage tolerance capability of the structure is demonstrated for an inspection interval, two missed inspections should be considered in the demonstration.

Limit load retention capability should be demonstrated after repeated-load cycling.

Environmental accountability should be included in the above demonstrations.

Limit load retention capability should be assessed with large understrength bonds and severe accidental damage present. Consider the two damage sources separately.

All tests should be on actual composite material being used. The alternative use of any other composite material should be substantiated.

Repair procedures may be part of the substantiation program and can be published in the Continued Airworthiness Section of the Maintenance Manual. This is not a regulatory requirement, but rather a highly desirable FAA goal.

All ultimate static tests on structures critical to flight safety should be conducted on full-scale component articles. Environmental effects should be taken into consideration.

All critical conditions should be tested to ultimate loads.

Guidance for Composite Aircraft Solar and Thermal Design Certification Criteria

The thermal environmental analysis should be based on a parametric study of the following data to identify the highest structural temperature:

Hour	Ambient Temperature	Solar Radiation
1100	111°F	330 Btu/ft^2 /hr
1200	114	355
1300	119	355
1400	122	330
1500	123	291
1600	124	231
1700	123	160

The above temperature values would not be exceeded 99.9 percent of the time, as derived from MIL-STD-210C statistical data. For the above data, the wind speed was 14 f.p.s., and the relative humidity was 3 percent.

The effect of cooling airflow may be taken into account. The FAA recommends the following:

After heat soak at the critical condition, the airplane taxis, takes off, and climbs to 1,000 feet above sea level. The airplane then accelerates in level flight to:

(1) The lesser of the design maneuvering speed, (V_A); or the aircraft operating speed limit (in § 91.117(b)) if maneuver loads are critical; or

(2) The lesser of the design cruise speed, (V_C); or the aircraft operating speed limit (in § 91.117(b)) if gust loads are critical.

In the case of a commuter category airplane, the design speed for maximum gust intensity, (V_B), applies instead of the design cruise speed, (V_C).

The aircraft operating speed limit in § 91.117(b) is 200 knots. This applies to major structure and may not be applicable to certain structures such as flaps and landing gear doors, which would be subject to limit loads at an earlier time in the flight profile. For a small airplane, a maximum taxi speed of 10 m.p.h. is recommended. A 4-minute taxi-time would be reasonable.

23.605 Fabrication methods (Amendment 23-23)

(a) The FAA found the process of paint removal by the Plastic Media Blasting method to show varying degrees of decrease in fatigue life and an increase in crack growth rate. Reports submitted by Battelle, Columbus Laboratories, indicate variations were experienced, depending on the material, material thickness, and number of plastic bead blastings. The FAA is concerned about the detrimental effects that plastic bead blasting could have on the static and fatigue strength of the base metal(s) when removing the paint or protective coating.

To approve a process specification for Plastic Media Blasting paint removal, it is necessary to establish that the method is not damaging to the aircraft; therefore, as a minimum, the following parameters should be considered:

- Material variations, for example, 7075-T6 alclad, 7075-T6 bare, 2024-T3 alclad, 2024-T3 bare, etcetera;
- Paint, primer, and number of coatings to be removed;
- Plastic media size and type;
- Plastic media hardness;
- Nozzle pressure (maximum);
- Distance (nozzle to component);
- Angular nozzle displacement;
- Plastic media (mesh) flow level and nozzle diameter; and
- Dwell time.

Plastic media vendor suppliers should demonstrate that their materials can be supplied to the same standards (shown above).

Coupon tests for static strength and fatigue properties (including crack initiation and propagation) should demonstrate the compatibility of the material and the non-plastic media blasted material.

Blasting equipment users should demonstrate that the equipment will give the proper pressures and precise metering of the plastic media flow rate. The media separator should be capable of removing foreign particles from the reclaimed media using methods, such as the following:

- Vibrating screens;
- Magnetic separation;
- Electrostatic separation;
- Floatation method;
- Liquid gravity settling;
- Wet classification;
- Dry screening;
- Air separators;
- Gravity separators;
- Fixed chamber separators; and
- Mechanical separators or cyclone classifiers.

(b) Weldwood Plastic Resin Glue is approved for wood spar construction in American Champion Aircraft Corporation (Bellanca) (Champion) (Aeronca) 7 and 8 series airplanes, specifically those listed in the following Aircraft Specification and Type Certificate Data Sheet:

Type Certificate A-759—Models 7AC, 7ACA, 7BCM, 7CCM, 7DC, 7EC, 7FC, 7GC, 7HC, 7JC, 7KC, 7ECA, 7GCAA, 7GCB, 7GCBC, and 7KCAB.

Type Certificate A-21CE—Models 8GCBC and 8KCAB.

See 23.603, Materials and workmanship, for information about composite materials.

Laminar flow technology is addressed in 23.21, Proof of compliance—General.

23.607 Fasteners (Amendment 23-48)

(a) Self-locking nuts, alone, should not be used in any system when movement of the joint may result in motion of the nut or bolt head relative to the surface against which it is bearing. Joint seizure (bearing, uniball, or bushing) does not have to be considered by this regulation when determining the relative motion of the parts in question, although it is advisable to do so. Suitable protection and material properties of the joint are required by 14 CFR Part 23, §§ 23.609 and 23.613.

Self-locking castellated nuts, with cotter pins or lockwire, may be used in any system.

Self-locking nuts should not be used with bolts or screws on turbine-engine airplanes in locations where the loose nut, bolt, washer or screw could fall, or be drawn into the engine air-intake scoop.

Self-locking nuts should not be used with bolts, screws or studs to attach access-panels or doors, or to assemble any parts that are routinely disassembled before or after each flight. This advice does not intend to exclude self-locking nut plates in these named applications. Nut plate designs permit the fastener to float, which is a desirable feature that is not provided by a non-floating fastener device.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

See AC 20-71, Dual-Locking Devices on Fasteners, for guidance about removable-fastener dual-locking devices for rotorcraft and transport category aircraft.

23.609 Protection of structure (Original)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

See 23.603, Materials and workmanship, for information that pertains to composite materials.

23.611 Accessibility (Amendment 23-48)

No policy available as of June 1, 1994.

23.613 Material strength properties and design values (Amendment 23-45)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

(d) No policy available as of June 1, 1994.

(e) Ideally, the structural test article (a whole wing, an empennage, a fuselage, etcetera) would contain all elements that are made of specification guaranteed-minimum-strength materials. Furthermore, each element's physical dimensions (geometry) would be at the nominal size, plus or minus specified tolerances, to conservatively represent the least strength or least stiff part that could be used according to approved design data (drawings, specifications, stress, or structural analyses).

Materials delivered according to specifications exceed the guaranteed-minimum-strength called out by drawings 99 times out of 100. Military Handbook Metallic Materials and Elements for Aerospace Vehicle Structures (MIL-HDBK-5), industry, and professional society material specifications intend this result, that is, with 95 percent confidence that 99 percent of the materials will exceed selected design values. That is, the materials used in the test article (and in production articles) are stronger than the minimum values certificated in the design.

Parts (elements) are manufactured and delivered to nominal sizes within tolerances. This means that they will either deliver minimal performance or more than promised.

There are, fundamentally, four actions an airplane designer can take to determine the strength of the airplane's structure:

(1) The designer can analyze the airplane structure to both limit and ultimate load conditions, using guaranteed minimum-strength-material properties and conservative geometric characteristics;

(2) The designer can test the airplane structure to limit loads and then analyze the airplane structure to ultimate loads;

(3) The designer can test the airplane structure to limit loads and, later, to ultimate conditions; and

(4) The designer can test the airplane structure to beyond ultimate load conditions.

This last option is usually chosen to determine excess strength, or growth capabilities. It also exceeds the minimum FAA airworthiness standards for which compliance should be shown. Certain airworthiness standards require one of these methods instead of the others.

Normally, any structural test article is expected to be stronger than the minimum "specification" design depicted by the drawings. The reason is 99 percent of the materials used to fabricate the airplane will normally exceed the material characteristics chosen for the design values (see the previous paragraph explanation). One common way to account for a test article that is expected to be stronger than the minimum design, identified on the drawings, is to increase the test article loads. A previous method was to "bump up" the test loads by a factor (at least 115 percent), which was employed during Civil Air Regulations (CAR) 04 and CAR 4a certifications of some airplane designs.

Alternatively, the airplane designer could choose to show that the strength properties of materials and dimensions of parts used in the structural component(s) tested are such that later components of these types—used in aircraft presented for certification—will have strength equal to or exceeding the strength of the components tested.

If in doubt about guaranteed-minimum-strength materials or conservative geometric characteristics in a design being assessed, do the following:

- Witness limit load tests without detrimental permanent deformation;
- Witness ultimate (ULT) load tests to 115 percent ULT without failures for 3 seconds; or
- Prepare static test reports that show how the test results were reduced to the minimum values represented in the design data.

See 23.603, Materials and workmanship, for additional information about composite or wood materials.

Other useful references: AC 20-33B, Technical Information Regarding Civil Aeronautics Manuals (CAM's) 1, 3, 4a, 4b, 5, 6, 7, 8, 9, 13 and 14; CAM 3, paragraphs 3.174-1 and 3.301-1; and CAM 4a, paragraph 4a.230.

Existing 14 CFR Part 23 rules related to the material correction factors are §§ 23.305(a) and (b); 23.307(a); 23.603(a); 23.613(c); and, before Amendment 23-45, § 23.615(a) and (c).

Policy: The intent of § 23.305, paragraphs (a) and (b), Strength and deformation requirements, § 23.307(a), Proof of structure standards, and § 23.603(a)(1), Materials and workmanship regulations, is that the lowest strength conforming airframe produced to a set of FAA-approved type design data will comply with the requirements of § 23.305.

23.615 Design properties (Amendment 23-45) [Removed]

23.619 Special factors (Amendment 23-7)

This table (see Figure 11) summarizes various special factors and where they are located. Read the appropriate airworthiness standards to determine when these special factors replace the factor of safety and when they multiply the factor of safety.

	§ 23.303	§ 23.365	§ 23.621	§ 23.623	§ 23.625
Factor of Safety	1.5				
Burst pressure factor		1.33, 1.67 with Special Conditions			
Casting factor			1-1.25 1.25-1.5 1.5-2.0 2.0-higher		
Bearing factor				FS = 6.67 FS ≥ 3.33 2.0≤FS≤3.33	
Fitting factor					1.15 1.33

FIGURE 11

See 23.603, Materials and workmanship, for information that pertains to composite materials.

23.621 Casting factors (Amendment 23-45)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) and (d) The following logic tree (Figure 12) permits a reader to quickly determine the inspection requirements associated with a chosen casting factor.

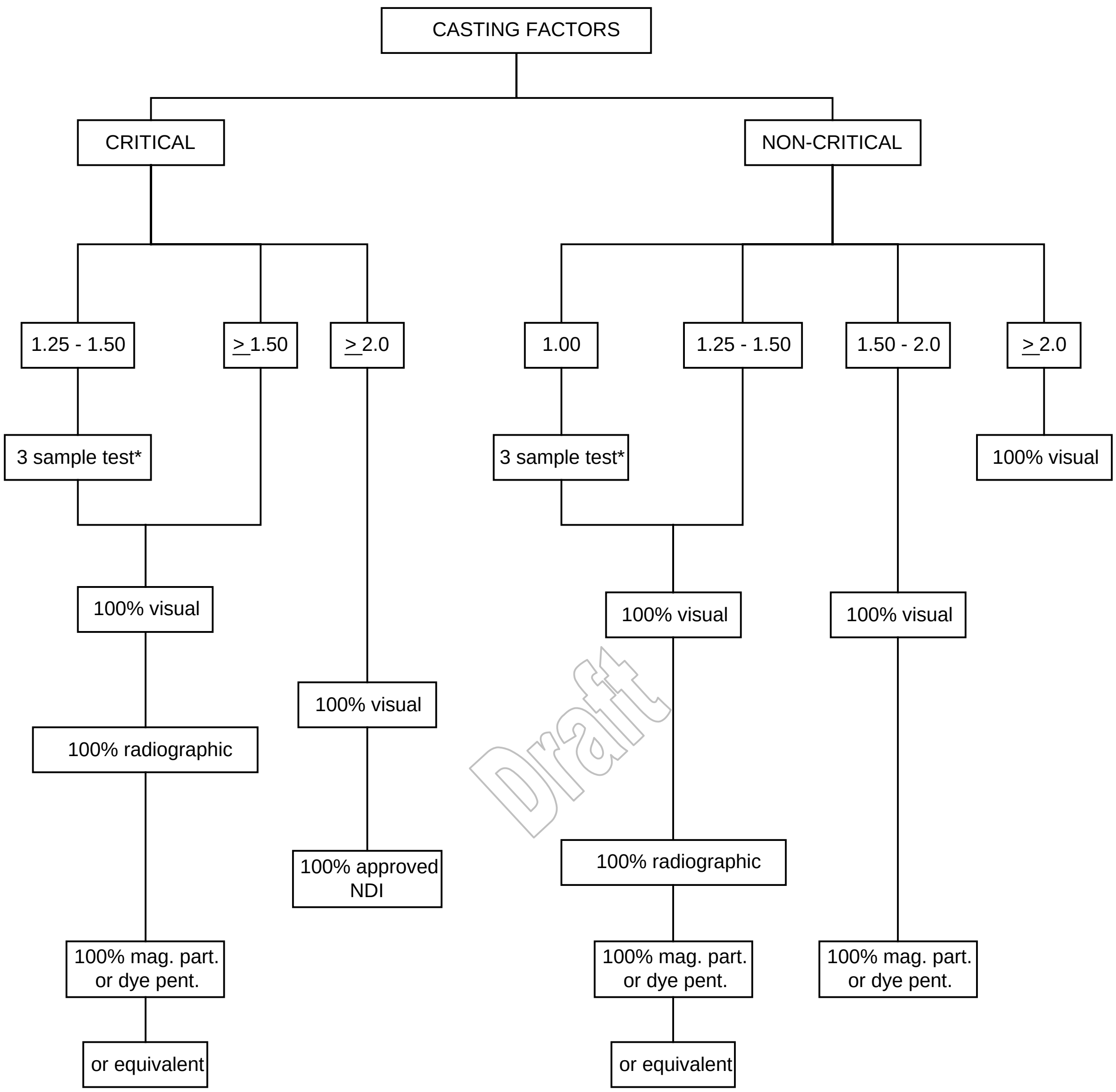

FIGURE 12

* Ultimate load corresponding to a casting factor of 1.25 (U = L x 1.5 x 1.25 = 1.875L); deformation requirements of § 23.305 at a Load = 1.15 x L, where U = ultimate and L = limit load.

(e) No policy available as of June 1, 1994.

23.623 Bearing factors (Amendment 23-7)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

23.625 Fitting factors (Amendment 23-7)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

(d) Question. Should the 1.33 fitting factor be applied to the dynamic emergency landing conditions as well as the static emergency landing conditions (see 14 CFR Part 23, §§ 23.562 and 23.561, respectively)?

Answer. Normally, the seat attachment fittings are included in the dynamic test and, therefore, there is no need to apply a fitting factor. If the restraint system attachment is separate from the seat and the attachment is not included in the dynamic test, the prescribed fitting factor should be applied to the attachment fittings.

23.627 Fatigue strength (Original)

Policy: Regarding the fatigue evaluation (under 14 CFR Part 23, § 23.627), the FAA has interpreted this standard as requiring only that the manufacturer exercise good design practice when avoiding severe stress concentrations, but not requiring a fatigue evaluation per se.

When the results of a fatigue test are plotted on an S-n diagram (stress versus number-of-cycles to failure), the fatigue limit is the constant stress level reached at a high number of cycles. Below that stress level, failure is not expected to occur. Aluminum alloys may not show a clearly defined fatigue limit. In such cases, the stress at 1×10^8 cycles is used to define an effective fatigue limit.

See AC 23-13, Fatigue and Fail-Safe Evaluation of Flight Structure and Pressurized Cabin for Part 23 Airplanes, for additional guidance about this airworthiness standard topic.

23.629 Flutter (Amendment 23-48)

(a) Flutter clearance, using rational analyses, has been required to 1.2 V_D since Amendment 23-7 became effective on September 14, 1969. Clearance, in terms of true or equivalent airspeed, obviously depends on whether appropriate density correction factors are included in the analysis. Before Amendment 23-7, including Civil Air Regulations (CAR) 3, it was required to show flutter-free operation to 1.0 V_D only.

The Simplified Flutter Prevention Criteria of Airframe and Equipment Engineering Report No. 45 (AEER 45), first published in Aviation Safety Engineering Release No. 330, dated December 2, 1949, defines criteria to establish flutter clearance to V_D for conventional airplane configurations.

Authorization to use AEER 45 (corrected February 1952)[1] as a means of meeting the flutter prevention requirements of CAR 3.311 first appeared in Civil Aeronautics Manual (CAM) 3.311-1 on March 13, 1952. The simplified criteria do not specify applicable airspeed or altitude limits.

CAM 3.311-1

No airspeed or altitude limits given

On December 1, 1978, Amendment 23-23 established an airspeed limit less than 260 knots equivalent airspeed (EAS) at altitudes below 14,000 feet and less than Mach 0.6 at altitudes at and above 14,000 feet.

Amendment 23-23

V_D < 260 EAS below 14,000 feet
and
M_D < Mach 0.6 at and above 14,000 feet

On January 8, 1979, AC 23.629-1, Means of Compliance with Section 23.169, “Flutter,” advised new lower airspeed limits for AEER 45[2] a design dive speed less than 200 m.p.h. EAS at altitudes below 14,000 feet.

AC 23.629-1

V_D < 200 m.p.h. EAS below 14,000 feet

[1] Existing copies of AEER 45 are undated and do not indicate if they are corrected. The correction appears on page 8 in the equation at paragraph 3(a). The corrected equation constant is 48.

On October 23, 1985, FAA published revised AC 23.629-1A and changed the airspeed limits for AEER 45 to a design dive speed less than 260 knots EAS at altitudes below 14,000 feet.

AC 23.629-1A

$V_D <$ 260 EAS below 14,000 feet

On September 7, 1993, Amendment 23-45 again changed the Mach number for AEER 45 to a design dive Mach number less than Mach 0.5 to closely agree with the calculated Mach number at 14,000 feet and 260 knots EAS.

Amendment 23-45

$V_D <$ 260 EAS
and
$M_D <$ Mach 0.5 at and above 14,000 feet

Note that 14 CFR Part 23 never established an altitude limitation on the applicability of AEER 45. Speed units, although not addressed in AEER 45, have historically been taken as EAS—except for wing torsional stiffness criteria that specify indicated airspeed (IAS) be used. However, at 14,000-feet altitude, the difference between EAS and IAS is small (about 3½ knots).

Finally, Amendment 23-48 made flight-flutter tests a requirement. Before February 9, 1996, the date of the amendment, freedom from flutter, control reversal, and divergence could be shown by either a rational analysis, by flight-flutter tests, or by simplified flutter prevention criteria.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

(d) Airframe and Equipment Engineering Report No. 45, “Simplified Flutter Prevention Criteria for Personal Type Aircraft,” by Robert Rosenbaum, and *Civil Aeronautics Manual 3, Supplement No. 11*, dated March 28, 1952, may be used for simple airplane designs to show compliance with flutter. Copies may be obtained by mail from the following: Manager, Standards Office (ACE-110), Federal Aviation Administration, Small Airplane Directorate, DOT Building, 901 Locust, Room 301, Kansas City, MO 64106.

(e) No policy available as of June 1, 1994.

(f) Policy: The objective is to prevent airplane flutter from occurring after the failure, malfunction, or disconnection of any single element in the primary flight-control system, in any tab-control system, or in any flutter damper. This can be achieved by balancing the control systems and then showing that the airplane is free from flutter. Alternatively, the objective can also be shown by doing the following:

- Incorporating a structural fail-safe design throughout the entire flight-control system and then demonstrating that the airplane is free from flutter; or

- Incorporating a combination of structural fail-safe designs and balanced-control system.

If a hinge pin single failure would allow the pin to fall out of the hinge, create hinge design features to prevent the pin from separating.

(g) No policy available as of June 1, 1994.

(h) The fail-safe design criterion of § 23.572(a)(2) is not an acceptable method of compliance for flutter. Section 23.629(f)(2) requires that the failure, malfunction, or disconnection of any single element be considered. Both control-surface balance and dual-load, path-tab-system designs have been judged as meeting the airworthiness requirement for irreversible systems.

When a dual-load path is chosen as a design option, one should design for 100 percent limit loads at all speeds up to the design dive speed, V_D/M_D. It should be possible to inspect the system. All elements of the system should also contain sufficient strength to withstand design loads between inspection intervals.

The fail-safe criterion in § 23.572(a)(2) imposes a static ultimate-load factor of 75 percent of the critical limit-load factor at design cruise speed, V_C. This criterion is inadequate for flutter substantiation of dual-load path primary-control systems, or tab-control systems because of the lower speeds and lower structural loads imposed.

(i) No policy available as of June 1, 1994.

See AC 23.629-1A, Means of Compliance with Section 23.629, “Flutter,” for additional information about this airworthiness standard topic.

WINGS

23.641 Proof of strength (Original)

No policy available as of June 1, 1994.

CONTROL SURFACES

23.651 Proof of strength (Original)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

23.655 Installation (Amendment 23-45)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

23.657 Hinges (Amendment 23-48)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

See 23.393, Loads parallel to the hinge line, for additional guidance about hinges.

Also**,** see 23.651, Proof of strength, for information that may affect hinges.

23.659 Mass balance (Original)

(a) No policy available as of June 1, 1994.

(b) No policy available as of June 1, 1994.

(c) No policy available as of June 1, 1994.

APPENDIX A TO PART 23—Simplified Design Load Criteria

A23.1(a) References:

1. Clousing, Lawrence A. and Turner, William N.: Flight Measurements of Horizontal Tail Loads on a Typical Propeller-Driven Pursuit Airplane During Stalled Pull-Outs at High Speed. RMR (WR A - 81), May 1944.

2. Matheny, Coyce E.: Comparison Between Calculated and Measured Loads on Wing and Horizontal Tail in Pull-Up Maneuvers. ARR L5H11 (WR L-193), Oct. 1945.

3. Garvin, John B.: Flight Measurements of Aerodynamic Loads on the Horizontal Tail Surface of a Fighter-Type Airplane. TN 1483, Nov. 1947.

4. Sadoff, Melvin and Clousing, Lawrence A.: Measurements of the Pressure Distribution on the Horizontal-Tail Surfaces of a Typical Propeller-Driven Pursuit Airplane in Flight. III—Tail Loads in Pull-Up Push-Down Maneuvers. TN 1539, Feb. 1948.

5. NACA Flight Research Maneuvers Section: Flight Studies of the Horizontal-Tail Loads Experienced by a Modern Fighter Airplane in Abrupt Maneuvers. Rept. 792, 1944.

The first paragraph of the Reference 1 CONCLUDING REMARKS reads:

> "With the test airplane operated within maneuvering limits which were considered safe by design specifications in use at the time the airplane was designed, units loads were measured on the stabilizer which were not only considerably in excess of the design unit loads, but which occurred in a direction opposite to the design loads."

Draft